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8. SUPPORTING TECHNICAL ANALYSES

8.1 Launch Vehicle Payload Capabilities

8.1.1 Gemini Launch Vehicle and Saturn I and IB - Payload capabilities are based on spacecraft injection into a low perigee elliptical catch-up orbit with apogee at the space station altitude. Rendezvous and orbit circularization is accomplished by onboard propulsion.

Weight-in-orbit capabilities used in performance evaluation of the various ferry and supply missions are presented in Figure 8.1-1. The current Gemini Launch Vehicle (GLV) - 3σ capability, is shown as a function of apogee altitude for orbits with an 87 nautical mile perigee. (Reference 8.1-1)

Saturn launch capabilities are for injection to 100 nautical mile perigee elliptic orbits. Payload capability to various apogee altitudes is calculated from the data of Reference 8.1-2, which, however, does not specify the associated injection probability.

8.1.2 GLV Payload Capability with Self-Injection - A substantial performance increase is attainable with GLV launched spacecraft by utilizing onboard propulsion for orbit injection. Payload for three self-inject system designs were evaluated using optimized launch trajectories for orbits with various apogee altitudes and a perigee altitude of 87 nautical miles. Present GLV heating, loads, and guidance constraints were observed.

Features of the three self-inject systems investigated and their associated payload increase are shown in Figure 8.1-2. The payload increase is obtained by deducting the weight of the self-inject propulsion system and additional attitude control and circularization propellant from the total incremental weight in orbit.

The quantity of propellant utilized in the two systems with Gemini thrusters is limited by the 524 seconds of allowable burning time of the thrusters. For the other system, however, the optimum propellant loading is usable since the 492

LAUNCH VEHICLE CAPABILITY vs APOGEE ALTITUDE

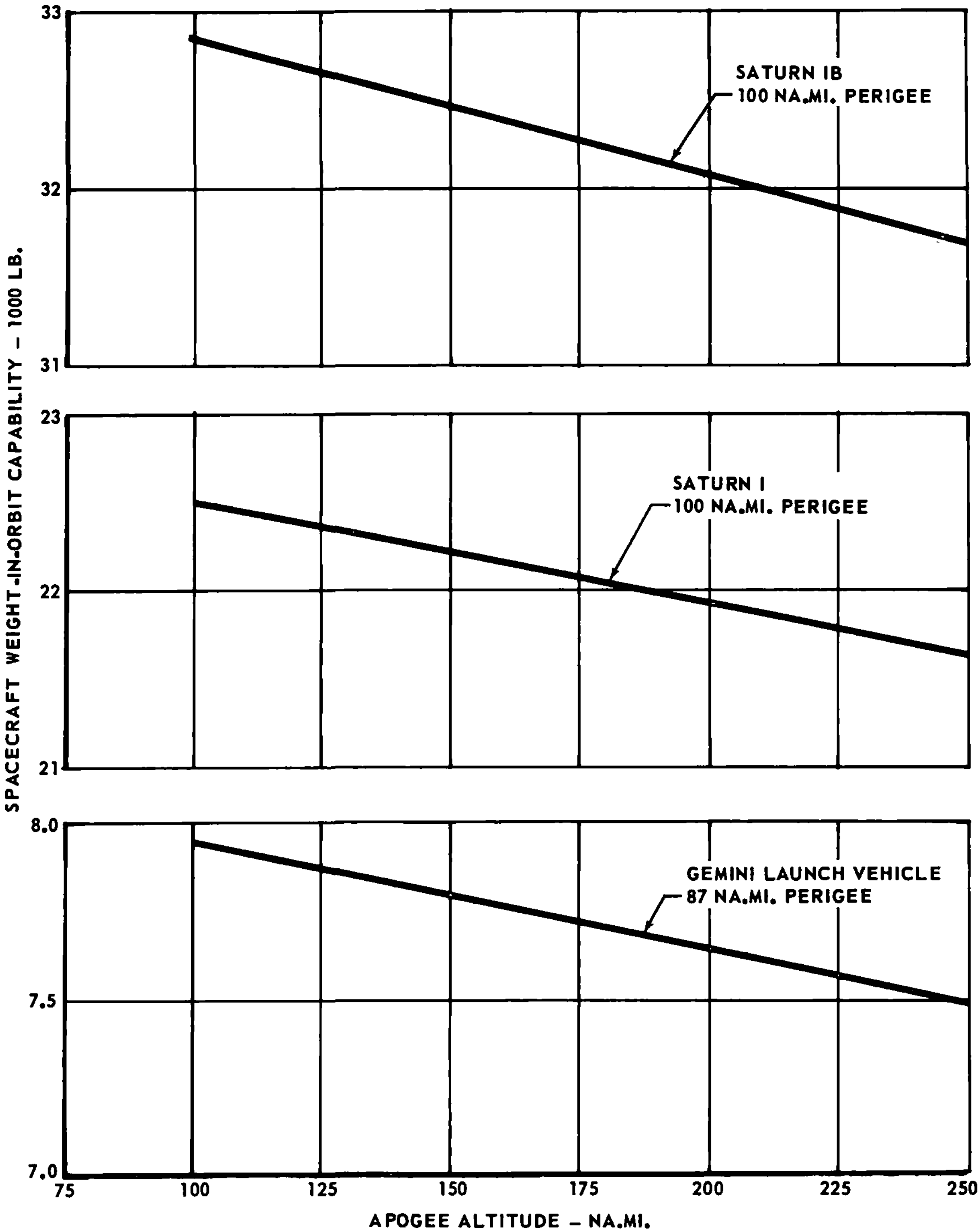


FIGURE 8.1-1

8.1.2 (Continued)

seconds of engine burning time is within the design capability of the engine.

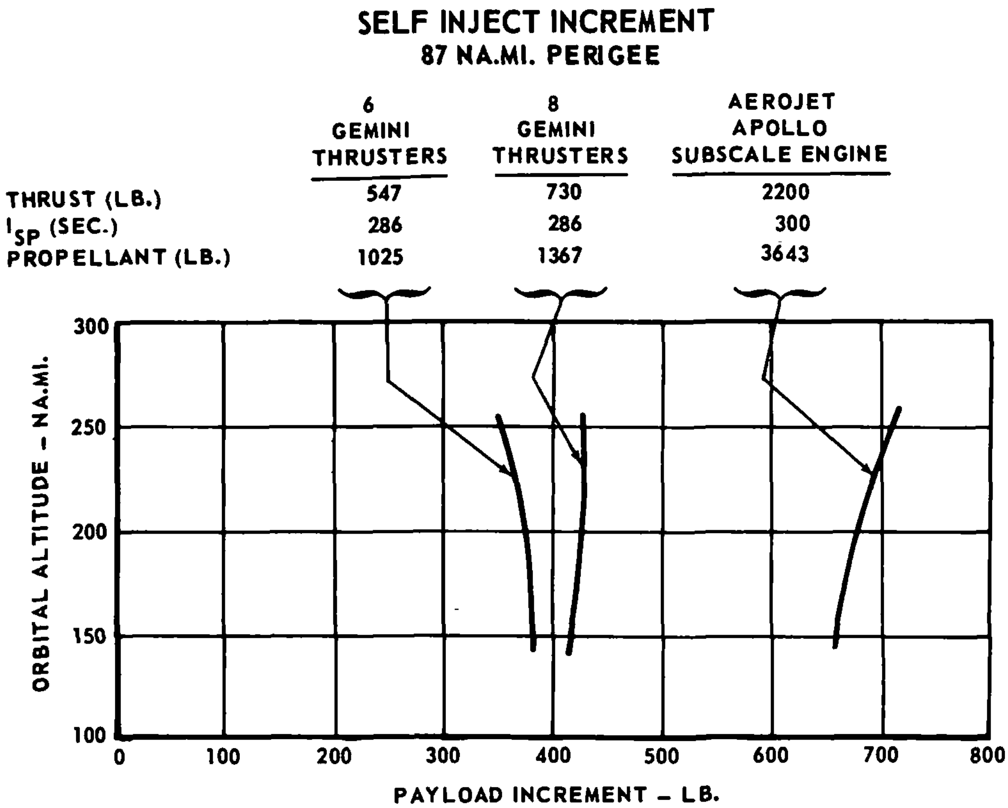


FIGURE 8.1-2

Range safety regulations forbid the impact of any stage on the African continent. The GLV second stage impact locations are shown in Figure 8.1-3 for a range of launch headings and 250 na. mi. apogee altitude. For launch headings near 90 degrees, only the largest self-inject system investigated assures a second stage impact point well clear of the African coast. For the smaller systems, impact will be on or near the African continent.

8.2 Structural Design Criteria and Loads

8.2.1 Structural Design Criteria - The structural criteria are based on those for Gemini (Reference 8.2-1) expanded to include unique requirements for the ferry and supply missions.

Design Weights - The design weights of the configurations are shown in Table 8.2-1. Where a range of weights is indicated the one producing the critical load is used for design.

GLV SECOND STAGE IMPACT POINT WITH SPACECRAFT SELF-INJECTION

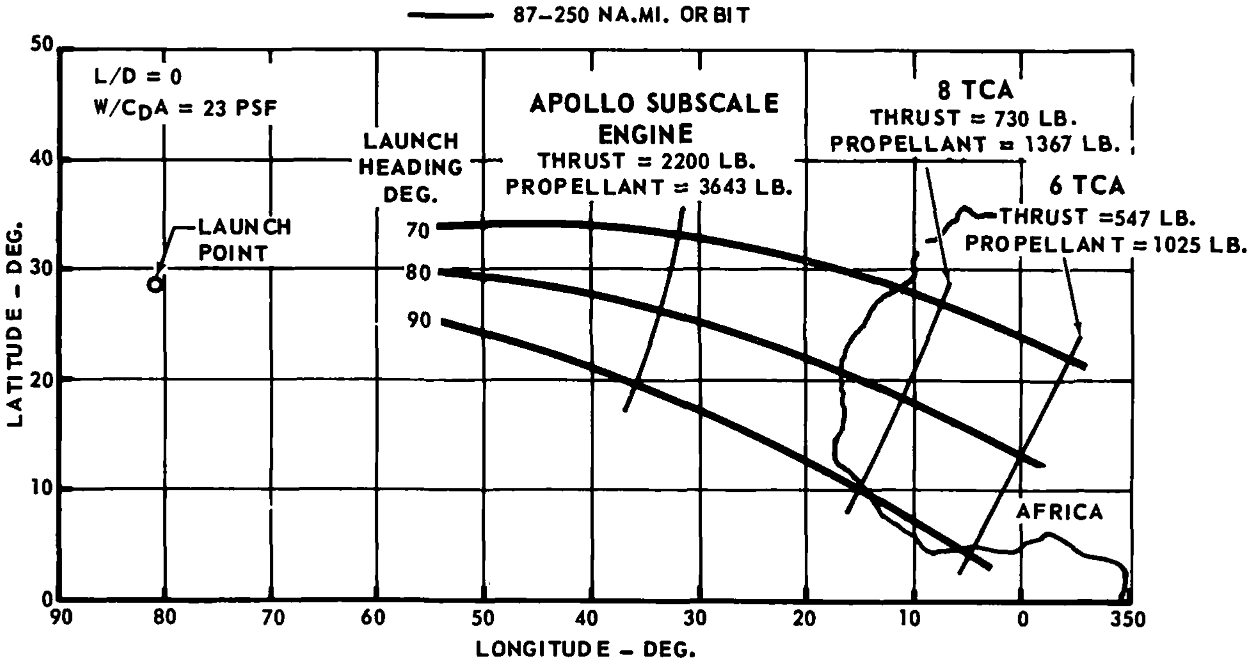


FIGURE 8.1–3

TABLE 8.2–1
DESIGN WEIGHTS

MISSION PHASE \ CONFIGURATIONS	WEIGHT – LB.			
	FERRY	FERRY-SUPPLY SATURN I	FERRY-SUPPLY SATURN IB	UNMANNED-SUPPLY
LAUNCH	6520–7520	21650	31700	12500–13500
RENDEZVOUS	6220–7220	20680	30800	7830–8830
ARTIFICIAL GRAVITY	6220–7220	20680 (MAX.)	30800 (MAX.)	3100–8830
RETROGRADE	4820–5820	4820–5820	4820–5820	
RE-ENTRY	4070–5070	4070–5070	4070–5070	
PARAGLIDER DEPLOYMENT	3650–4650	3650–4650	3650–4650	

8.2.1 (Continued)

Factors of Safety - Ultimate loads are 1.36 times limit loads for all configurations except the unmanned supply for which the factor of safety is 1.25.

The ultimate temperature is the greater of: (1) limit temperature increased by 200°F for re-entry and by 100°F for launch or (2) temperature resulting from a 15% increase in calculated heating rate.

Ultimate temperatures are combined with limit loads or ultimate loads combined with limit temperatures, whichever is critical.

Docking - The structure shall accommodate docking impact resulting from combination of the following conditions:

Maximum Relative Longitudinal Velocity	1.5 fps
Maximum Relative Lateral Velocity	±.5 fps
Maximum Relative Rolling Velocity	±10.0 degrees per second
Maximum Pitching or Yawing Rates	±.75 degrees per second
Maximum Lateral Mismatch Distance	±1.0 feet
Maximum Mismatch Attitude	±10.0 degrees

Mooring - A 1 g artificial gravity load factor shall be used for design purposes.

Meteoroids - Whipple's 1961 meteoroid distribution and Bjork's penetration equation shall be used for analysis. (Reference 8.2-2)

8.2.2 Ferry Spacecraft Launch Loads - Since the Ferry configuration is the same as Gemini, current Gemini loads are used. (Reference 8.2-3)

8.2.3 Ferry/Supply Spacecraft Launch Loads - These spacecraft consist of a Gemini Ferry with a supply adapter launched on Saturn I or IB. Aerodynamic data are well defined for the Ferry. Loads over the supply adapter are obtained using derived aerodynamic data. The pitching acceleration is zero at maximum q design conditions. This assumes some engine thrust deflection to balance aerodynamic moments. The maximum dynamic pressure and other trajectory data were obtained from References 8.2-4 and 8.2-5.

8.2.3 (Continued)

The maximum q condition produces the largest loads for both Saturn I and Saturn IB launches. The loads determined from the Saturn I trajectory are greater than for the Saturn IB trajectory due to the greater maximum dynamic pressure. Therefore, only Saturn I maximum q loadings are shown.

The burnout pitching accelerations are calculated using appropriate thrust moments and space vehicle pitching moment of inertia data. Stage burnout conditions assume maximum engine gimbal deflection and the dynamic pressure is assumed to be negligible.

Although the first stage burnout loads are less than at maximum q, they occur at elevated temperatures. The maximum burnout loads occur for Saturn IB launch trajectories and therefore only these are presented. The temperatures corresponding to these loading conditions are somewhat less than Gemini.

Cherokee Rocket Escape Configuration - A Ferry/Supply Spacecraft with an escape system which includes 8 Cherokee rockets and 4 fins attached to the adapter was investigated. Launch loads for this configuration are shown in Table 8.2-2.

The Saturn I launch produces the largest maximum q loads, and the Saturn IB launch produces the largest stage burnout loads.

The maximum q normal launch loads for the Cherokee escape system configuration are slightly lower than Gemini spacecraft loads due to lower dynamic pressure experienced during the trajectory. The stage burnout loads applied to the Ferry are considerably less than on Gemini because of lower accelerations.

Escape Loads with Cherokee Rocket Escape System - The escape loads presented in Table 8.2-3 are for the instant the spacecraft separates from the launch vehicle with the escape rockets burning. The maximum αq during escape is assumed to be the same as the design launch maximum αq.

Escape Tower Configuration - Another escape system utilizes a tower similar

TABLE 8.2-2

ULTIMATE LAUNCH LOADS

FERRY/SUPPLY SPACECRAFT
CHEROKEE ROCKET ESCAPE CONFIGURATION

CONFIGURATION	CONDITION	LOAD SHEAR LB. MOMENT IN.-LB. AXIAL LB.	GEMINI BODY STATION			
			103.44	13.44	-36.56	-138.56
SATURN I LAUNCH VEHICLE LAUNCH PAYLOAD OF 22,500 LB.	MAXIMUM q $q = 690$ PSF $\alpha = 14°$ $N_n = .6$ $N_a = 1.62$ $\ddot{\theta} = 0.0$	SHEAR MOMENT AXIAL	11,800 570,000 30,450	37,540 2,737,220 39,720	38,740 4,644,220 40,320	51,840 9,003,700 80,220
SATURN IB LAUNCH VEHICLE LAUNCH PAYLOAD OF 32,500 LB.	STAGE BURNOUT $N_n = .276$ $N_a = 3.97$ $\ddot{\theta} = -.364$ RAD/SEC^2	SHEAR MOMENT AXIAL	4,710 169,500 27,000	7,670 725,000 45,570	7,300 1,087,000 46,670	23,450 2,165,000 181,670

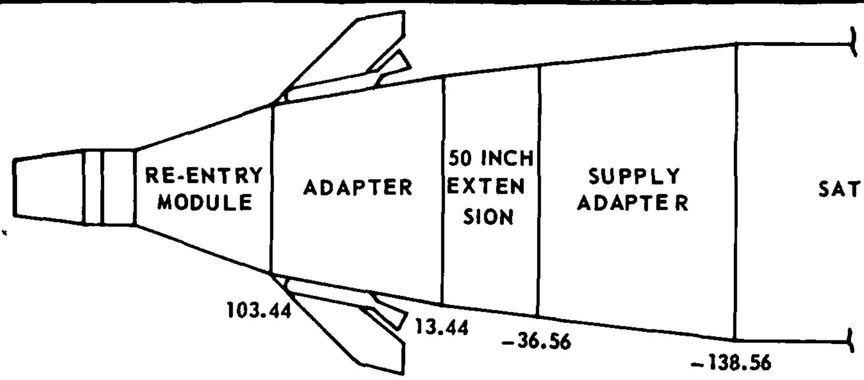


TABLE 8.2-3

ULTIMATE ESCAPE LOADS

FERRY/SUPPLY SPACECRAFT
ADAPTER ROCKET CONFIGURATION

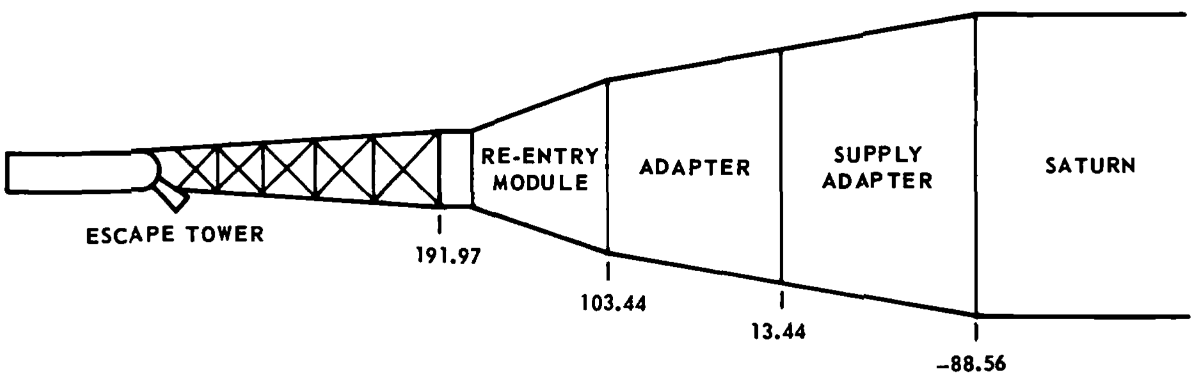
CONFIGURATION	LOAD	GEMINI BODY STATION 103.44 CAPSULE/RETRO SECTION
SATURN I LAUNCH VEHICLE MAXIMUM q CONDITION	SHEAR LB. MOMENT IN-LB. AXIAL LB.	11,600 287,000 87,500

8.2.3 (Continued)

to that used for Project Mercury. Scaled up Mercury tower aerodynamic data was used to determine design loads. Four hundred pounds of ballast are placed on the forward end of the rocket. Table 8.2-4 presents design launch loads for this configuration.

TABLE 8.2-4
ULTIMATE LAUNCH LOADS
FERRY/SUPPLY SPACECRAFT ESCAPE TOWER CONFIGURATION

			GEMINI BODY STATIONS			
CONFIGURATION	CONDITION	LOAD SHEAR LB. MOMENT IN.-LB. AXIAL LB.	191.97	103.44	13.44	-88.56
SATURN I LAUNCH VEHICLE LAUNCH PAYLOAD 22,500 LB.	MAXIMUM q $q = 690$ PSF $\alpha = 14$ $N_n = .6$ $N_a = 1.62$ $\ddot{\theta} = 0.0$	SHEAR MOMENT AXIAL	6,960 618,000 20,200	16,660 1,568,000 43,000	34,060 3,650,500 51,000	47,160 7,631,000 91,500
SATURN IB LAUNCH VEHICLE LAUNCH PAYLOAD 32,500 LB.	STAGE BURNOUT $N_n = .276$ $N_a = 3.97$ $\ddot{\theta} = -.364$ RAD/SEC^2	SHEAR MOMENT AXIAL	2,700 485,000 12,420	6,280 693,000 34,850	8,260 1,288,000 48,500	24,200 2,370,000 183,500



The bending moments at Gemini Station 103.44 are over twice the present Gemini moment. The bending moments at Gemini Station 13.44 are 10 to 30 percent larger than present Gemini. The launch loads at the tower attach point (Gemini Station 191.97) are approximately 10 times the current Gemini maximum dynamic pressure loads at this station.

8.2.3 (Continued)

Escape Loads with Tower Escape System - Design escape loads for this configuration were estimated by assuming that critical conditions would exist at points in the escape maneuver and at angles of attack which resulted in critical loads for Mercury. Table 8.2-5 defines four such conditions and presents the corresponding design loads.

TABLE 8.2-5
ULTIMATE ESCAPE LOADS
FERRY/SUPPLY SPACECRAFT ESCAPE TOWER CONFIGURATION

CONFIGURATION	CONDITION	LOAD	GEMINI BODY STATION 191.97 TOWER ATTACH POINT
SATURN I LAUNCH VEHICLE	1. $\alpha = 10°$ NO THRUST MAXIMUM WT.	MOMENT IN.-LB. AXIAL LB.	719,000 12,980 COMPRESSION
MAXIMUM q 150-INCH TOWER	2. $\alpha = 10°$ THRUST MAXIMUM WT.	MOMENT IN.-LB. AXIAL LB.	719,000 101,500 TENSION
400 LB. BALLAST	3. $\alpha = 50°$ THRUST MINIMUM WT.	MOMENT IN.-LB. AXIAL LB.	440,000 106,000 TENSION
	4. $\alpha = 50°$ NO THRUST MINIMUM WT.	MOMENT IN.-LB. AXIAL LB.	440,000 10,070 COMPRESSION

8.2.4 Unmanned Supply Spacecraft Launch Loads - Design loads are calculated for the two Unmanned Supply Spacecraft launched by the Gemini Launch Vehicle (GLV). Gemini aerodynamic data, adjusted for the changes in external configurations, is used.

The temperatures on these two spacecraft are higher than on Gemini due to a lower surface emissivity used to obtain a greater degree of passive temperature control in orbit.

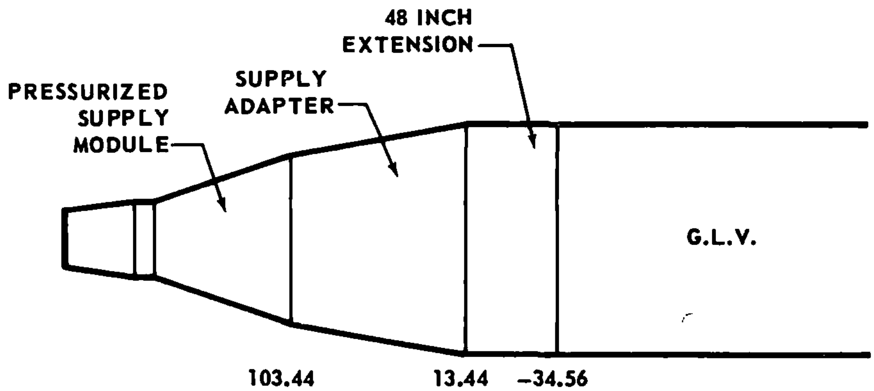
Stripped Gemini - Table 8.2-6 presents design launch loads. The Stripped Gemini has the Re-entry Control System section removed which reduces the aerodynamic loads and this, combined with the lower ultimate safety factor, results in lower

8.2.4 (Continued)

ultimate loads than Gemini. The GLV loads at Titan Station 276.825 (adapter-launch vehicle interface), however, are approximately 30 percent higher than with present Gemini due to a 48 inch skirt added to enclose the self-inject engine.

TABLE 8.2-6
ULTIMATE LAUNCH LOADS
UNMANNED SUPPLY SPACECRAFT-STRIPPED GEMINI

CONDITIONS	LOADS SHEAR LB. MOMENT IN.-LB. AXIAL LB.	GEMINI BODY STATIONS 103.44	13.44	-34.56
MAXIMUM q				
$q = 820$ PSF	SHEAR	13,600	27,400	31,680
$\alpha = 14°$				
$N_a = .6$	MOMENT	544,000	2,205,000	3,622,200
$N_a = 2.0$				
$\ddot{\theta} = 0.0$	AXIAL	26,200	51,300	52,000
STAGE BURNOUT				
$N_n = .5$	SHEAR	5200	14,350	14,550
$N_a = 5.2$	MOMENT	208,000	872,000	1,560,000
$\ddot{\theta} = -2.0$ RAD/SEC.^{2}	AXIAL	21,500	82,800	84,000

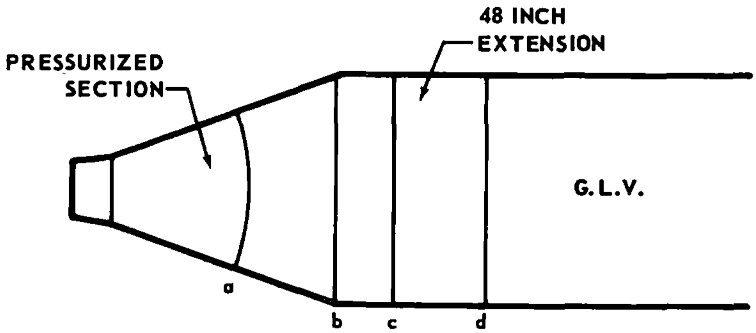


Specific Design - The design launch loads are shown in Table 8.2-7. The launch vehicle spacecraft interface loads are 40% larger than those occurring with Gemini.

8.2.4 (Continued)

TABLE 8.2-7
ULTIMATE LAUNCH LOADS
UNMANNED SUPPLY SPACECRAFT-SPECIFIC DESIGN

CONDITIONS	LOADS SHEAR-LB. MOMENT IN.-LB. AXIAL-LB.	SPECIFIC MODULE BODY STATION			
		a BASE OF PRESSURIZED SECTION	b BASE OF CONICAL SECTION	c BASE OF MODULE	d MODULE/BOOSTER INTERFACE TITAN STA. 276.825
MAXIMUM q $q = 820$ PSF $\alpha = 14^{\circ}$ $N_n = .6$ $N_a = 2.0$ $\ddot{\theta} = 0.0$	SHEAR MOMENT AXIAL	15,800 655,000 20,700	28,800 1,643,000 42,000	26,800 2,640,000 53,750	29,000 3,940,000 56,250
STAGE BURNOUT $N_n = .5$ $N_a = 5.2$ $\ddot{\theta} = 2.0$ RAD/SEC.^{2}	SHEAR MOMENT AXIAL	3300 89,200 14,750	9750 348,000 54,000	13,450 690,000 81,200	13,650 1,313,000 84,500

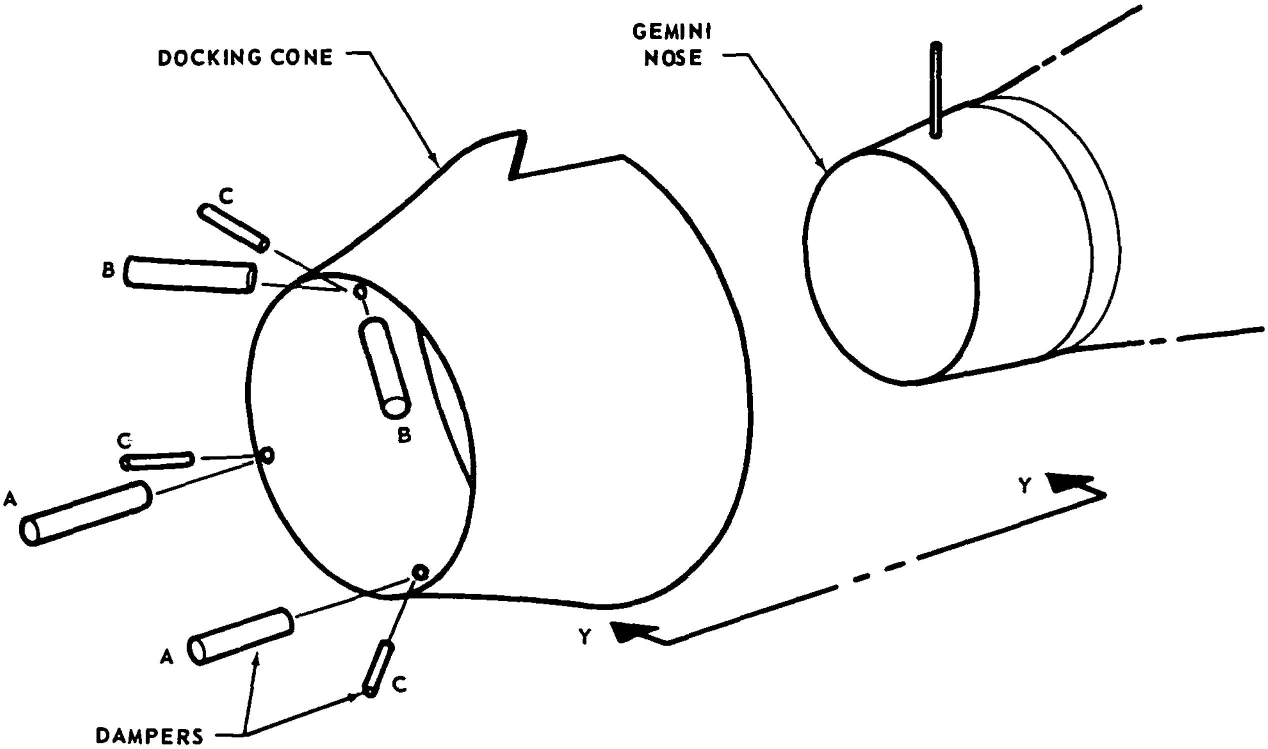


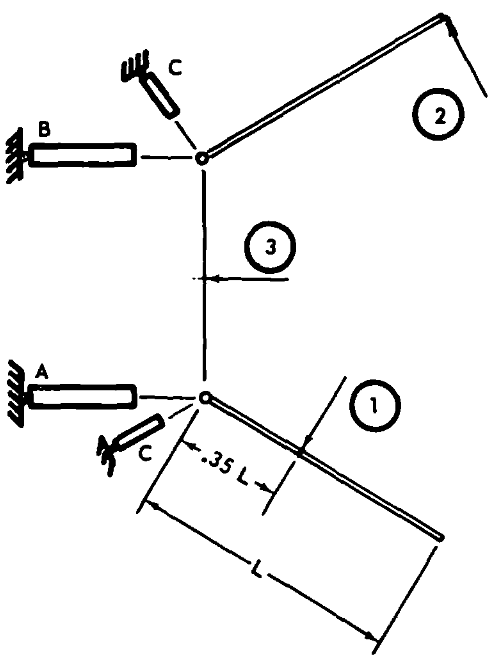
8.2.5 Docking Loads

Nose Docking - A parametric investigation of loads resulting from nose docking of the Ferry and Ferry/Supply spacecraft to MORL was made based on criteria discussed in Section 8.2.1. Loads for initial impact on either side of the conical surface of the docking cone, and on the stops at the bottom of the cone, were studied to determine design conditions and docking cone stroke requirements.

A schematic arrangement of the docking cone, including location of the dampers is shown in Figure 8.2-1. Associated with the three initial impact locations shown are three load conditions: Condition (1) for which maximum bending and shear loads

NOSE DOCKING SYSTEM





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EXISTING GEMINI VALUES – LONGITUDINAL DAMPERS

(A) $C = 1.45$ LB-SEC^2/IN.^{2}
$K = 25$ LB/IN.

(B) $C = 1.92$ LB-SEC^2/IN.^{2}
$K = 25$ LB/IN.

RADIAL DAMPERS

(C) $C = 1.66$ LB-SEC^2/IN.^{2}

FIGURE 8.2-1

8.2.5 (Continued)

are applied to the spacecraft, Condition (2) which produces maximum stroke requirements, and Condition (3) for which the maximum longitudinal load is applied to the spacecraft.

Twelve different space station configurations were studied to evaluate the effect of station mass, moment of inertia and center of gravity location on load and stroke. Two station configurations which bracket the assumed range of feasible MORL arrangements and impose the most severe design requirements on the spacecraft and the docking mechanism are:

A. The largest mass station configuration considered weighs 94,700 lb. and has a moment of inertia of 1,633,000 slug ft^2. It consists of a Saturn IB launched basic MORL with the inert SIVB stage attached, a Saturn IB launched cargo module and two nose dock-nose moor Ferry Spacecraft moored in diametrically opposite positions. The center of gravity is 7.2 inches from the docking port centerline.

B. The smallest mass station configuration considered weighs 21,650 lb. and has a moment of inertia of 48,300 slug ft^2. It consists of a Saturn I launched basic MORL only. The center of gravity is 78 inches from the docking port.

The loads applied to the spacecraft and the damping coefficient and spring constants needed to limit the stroke to the Agena value of 6 inches are shown in Table 8.2-8.

TABLE 8.2-8
NOSE DOCKING PARAMETERS

ITEM	FERRY SPACECRAFT	FERRY/SUPPLY SPACECRAFT
LONGITUDINAL LOAD FACTOR – g's	.635	.518
STROKE – INCHES	6.0	6.0
LONGITUDINAL DAMPING COEF. – LB.-SEC.^{2}/IN.^{2}	3.82	13.2
LONGITUDINAL SPRING CONSTANT – LB./IN.	66	228
LIMIT LOAD – LB.	2800	9700

8.2.5 (Continued)

The method of analysis is programmed on the IBM 1620 and involves an "instant center" calculation of the docking cone upon initial impact by the Ferry Spacecraft.

For a given docking cone energy absorber stiffness (damping coefficient, C, $\frac{\#\text{-sec}^2}{\text{in}^2}$), the maximum load resulting from a symmetrical impact, Condition (3) is significantly greater than loads resulting from Condition (1) as shown on Figure 8.2-2. The allowable strength envelope, Figure 7.3-1 , indicates that Condition (1), however, is the designing case since considerably more strength exists for axial compression loads than for bending loads. The load applied to the spacecraft for Condition (1) versus stroke is shown in Figure 8.2-3. Stroke requirements are determined by an initial impact on the top of the cone (Condition (2)) which in turn determines the amount of damping necessary. The stroke also includes that necessary to stop relative rotation of the spacecraft and space station after lock-on.

A docking mechanism designed for Ferry Spacecraft requirements will also accommodate the Unmanned Supply Spacecraft. Since the Unmanned Supply Spacecraft probably will dock on the end of the space station, relative rotations will be much less than for Ferry configurations.

Aft Docking - The aft docking mechanism consists of four 90° forks symetrically mounted on the station and a docking ring located on the rendezvous spacecraft which engages the forks in Figure 3.3-2. Compression springs on each fork absorb the impact energy. Maximum loads on the rendezvous spacecraft occur for symmetrical impact on all four forks. Maximum stroke requirements of the system are determined by initial impact on one fork. Design criteria for aft docking is the same as for nose docking with the exception that relative impact velocity for contact at the apex of the forks is 2.3 ft./sec. This velocity results from a 1.5 ft./sec. initial velocity at the tip of the forks and acceleration at 1.0 ft./sec.^{2} until the apex is contacted. Design loads for aft docking of Ferry and Ferry/Supply

INITIAL DOCKING LOAD
vs.
LONGITUDINAL DAMPING COEFFICIENT

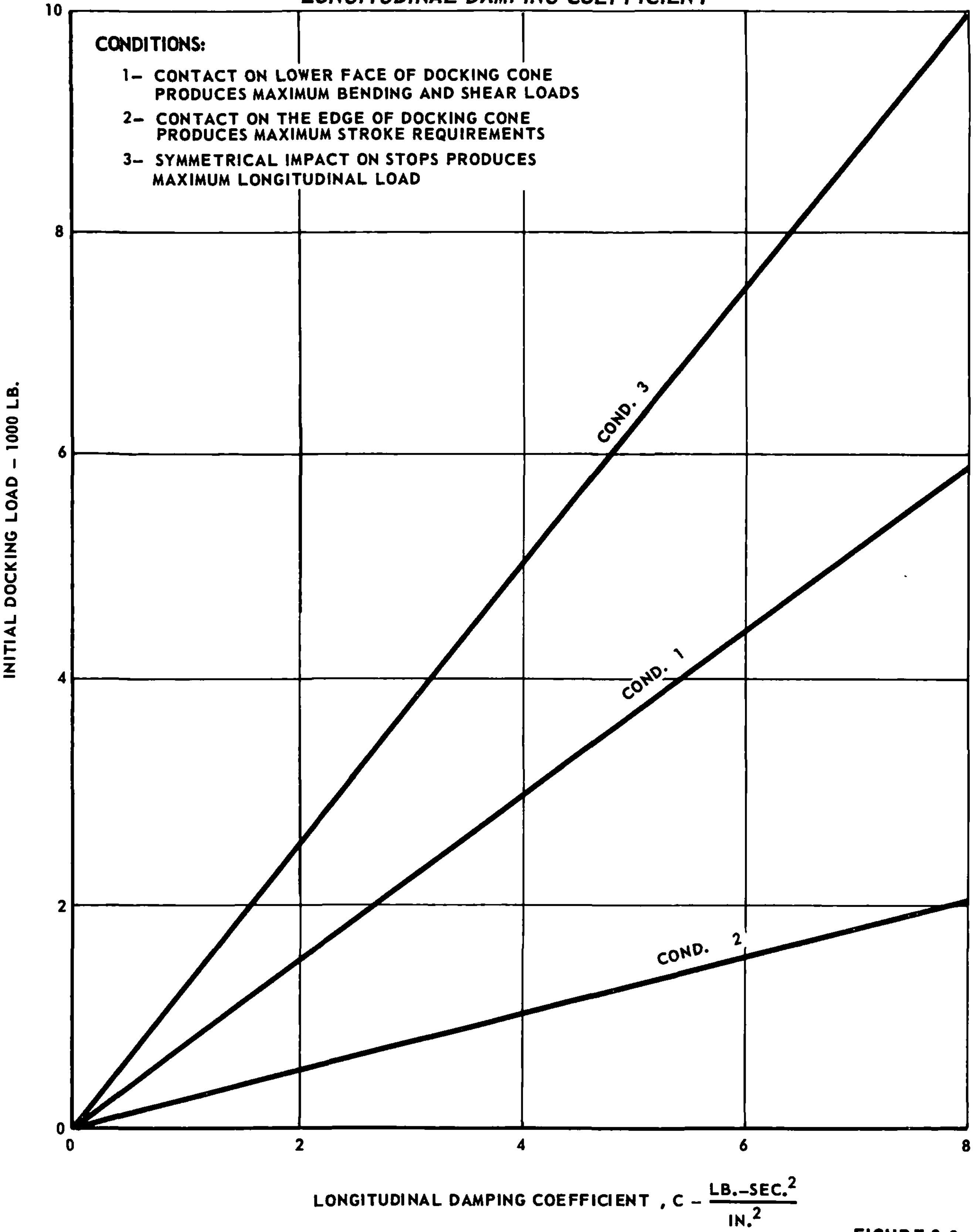


FIGURE 8.2–2

NOSE DOCKING LOADS vs AVAILABLE SYSTEM STROKE

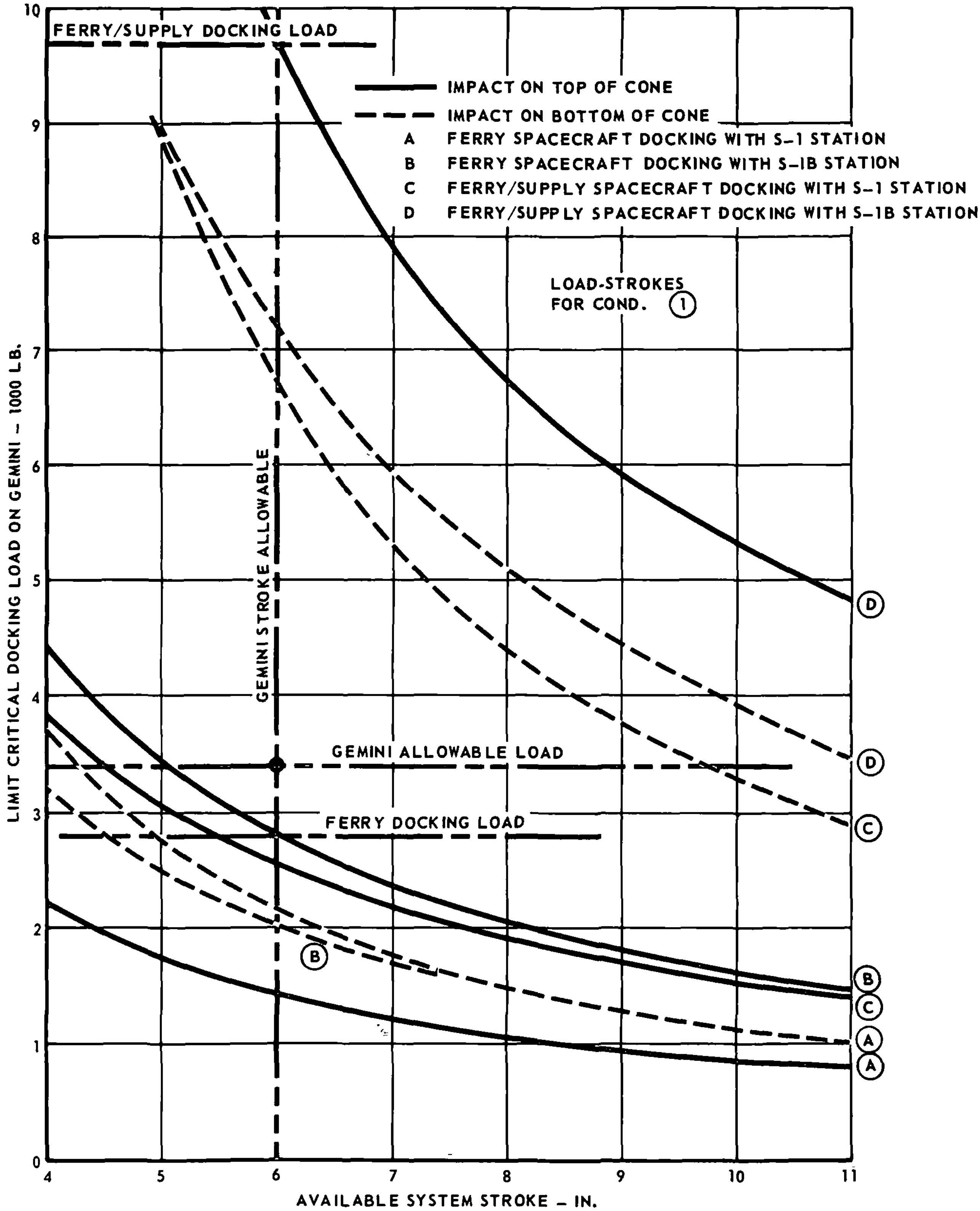


FIGURE 8.2-3

8.2.5 (Continued)

spacecraft, for the same maximum load factors as for nose docking, are shown in Table 8.2-9.

TABLE 8.2-9

AFT DOCKING PARAMETERS

ITEM	FERRY SPACECRAFT	FERRY/SUPPLY SPACECRAFT
LONGITUDINAL LOAD FACTOR – g's	.635	.518
MAXIMUM LOAD PER FORK – LB.	2380	8280
LONGITUDINAL SPRING CONSTANT – LB./IN.	417	1120
STROKE – IN.	5.8	6.5

8.2.6 Mooring Loads - Loads encountered while moored result from artificial gravity operation and, where applicable, from internal pressure loads which tend to separate the moored spacecraft from the MORL.

Design limit loads for 1 g artificial gravity operation are presented in Tables 8.2-10 (aft moored) and 8.2-11 (nose moored). The 1.36 factor of safety applies since failure of a mooring connection would endanger the station crew.

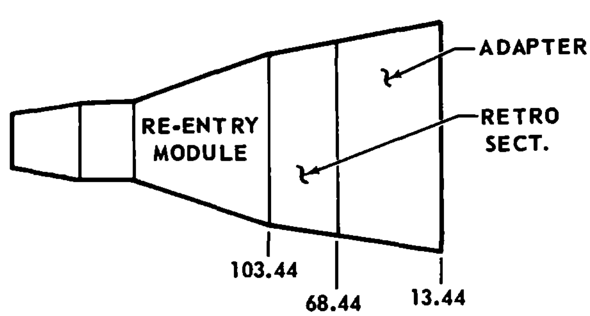
Pressurization loads are based on an ultimate burst pressure of 12 psi.

TABLE 8.2-10

LIMIT MOORING LOADS

AFT MOORED FERRY SPACECRAFT

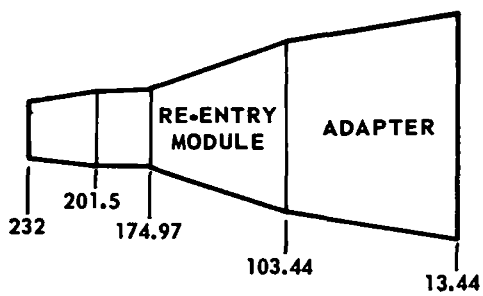
LOAD	UNIT	GEMINI BODY STATION		
		103.44	68.44	13.44
SHEAR	LB.	4800	6700	7200
MOMENT	IN.-LB.	170,000	361,000	705,000
AXIAL	LB.	4800	6700	7200



8.2.6 (Continued)

TABLE 8.2-11
LIMIT MOORING LOADS
NOSE MOORED FERRY SPACECRAFT

LOAD	UNIT	GEMINI BODY STATION				
		232	201.5	173.97	103.44	13.44
SHEAR	LB.	7200	6780	6250	2400	0
MOMENT	IN.-LB.	860,000	647,000	465,000	105,000	0
AXIAL	LB.	7200	6780	6250	2400	0



8.2.7 Acoustical Loads - The primary sources of noise are the launch vehicle engines and the turbulent boundary layer. An analysis of noise levels for the Ferry/Supply Spacecraft at lift-off and at maximum dynamic pressure for Saturn I or Saturn IB launches, indicates that the noise levels are less than encountered on the Gemini Launch Vehicle. The maximum acoustical loads, therefore, are less than on Gemini, and no redesign appears necessary.

8.3 Rendezvous Analysis - The Ferry Spacecraft performs rendezvous with the MORL at an altitude greater than the 161 na. mi. for Gemini-Agena. To determine necessary changes from Gemini methods, rendezvous was investigated with the MORL in a 250 na. mi. orbit for (A) use of Gemini rendezvous techniques, (B) use of pre-rendezvous in-orbit correction of injection errors, and (C) decrease of the design out-of-plane angle (with respect to the station orbit) at Ferry injection from the Gemini value of 0.53°.

From these studies, which are described in subsequent paragraphs, it is concluded that use of pre-rendezvous in-orbit correction of injection errors allows rendezvous and docking with 700 fps (ΔV) maneuver capability.

8.3 (Continued)

8.3.1 Gemini Rendezvous Technique - The following is a brief outline of the Gemini rendezvous technique - (for a complete description see Reference 8.3-1 and 8.3-2): (1) Gemini is nominally injected into orbit at perigee and at nearly maximum out-of-plane position. (2) The platform is slaved to local vertical, radar lock-on acquired, and rendezvous calculations are made. (3) At approximately first apogee the initial rendezvous impulse is applied. (4) After four small corrective impulses, and 270° additional travel in orbit, the rendezvous is completed.

With this procedure, the initial rendezvous pulse is applied at nearly maximum out-of-plane position and it nearly circularizes the orbit and nulls velocity, existing at that time, of the Ferry normal to the station orbit plane. The terminal impulse of the rendezvous is proportional to the distance the Ferry is out of the station orbit plane at the time the initial impulse is applied.

Injection dispersions, which are corrected during the closed-loop phase of rendezvous, have a large effect on ΔV requirements. Use of the methods of Reference 8.3-3 for the case of a 250 na. mi. orbit results in a ΔV requirement of 839 fps, including 50 fps for the four corrective impulses and 50 fps for docking, for 99.7% (3σ) of injection conditions, as compared to 615 fps for the nominal case. The calculated 3σ ΔV is greater than the existing 700 fps capability of Gemini.

As a result of this analysis, methods of reducing the ΔV required for rendezvous were investigated.

8.3.2 Pre-Rendezvous In-Orbit Correction of Injection Errors - A promising method of reducing ΔV requirements is to make in-orbit corrections of the injection dispersions prior to commencing rendezvous. These corrections are made at positions in orbit more favorable than for correction during the closed-loop rendezvous phase thus bringing the ΔV needed closer to the nominal value. The procedure is to inject and allow the first two orbits for correction of: (A) apogee altitude

8.3.2 (Continued)

error at the perigee one orbit after injection, and (B) in-plane angular dispersion at the following apogee. For an on-time launch, rendezvous is completed in 3-1/4 orbits. If launch were delayed, correction (A) would be delayed until one and one half orbits before initiating rendezvous in order to use tracking data acquired during the catch-up phase. Corrections are computed on the ground from injection measurements and are relayed to the Ferry. The method relies on the fact that injection errors can usually be measured by approximately an order of magnitude more accurately than they can be controlled.

Analysis, based on the measurement errors given in Table 8.3-1, shows that the dispersions at initiation of rendezvous are reduced to about one fifth of those obtained when no pre-rendezvous in-orbit compensations of injection errors are made (Figure 8.3-1).

TABLE 8.3-1

ASSUMED INJECTION MEASUREMENT ACCURACIES

VELOCITY	±3.2 FT./SEC.
FLIGHT PATH ANGLE	±0.04 DEG.
HEADING	±0.01 DEG.
VERTICAL DISPLACEMENT	±300 FT.
OUT OF PLANE DISPLACEMENT	±200 FT.
DOWNRANGE DISPLACEMENT	NEGLIGIBLE
TIME	0

POSITION DISPERSION ABOUT NOMINAL APOGEE

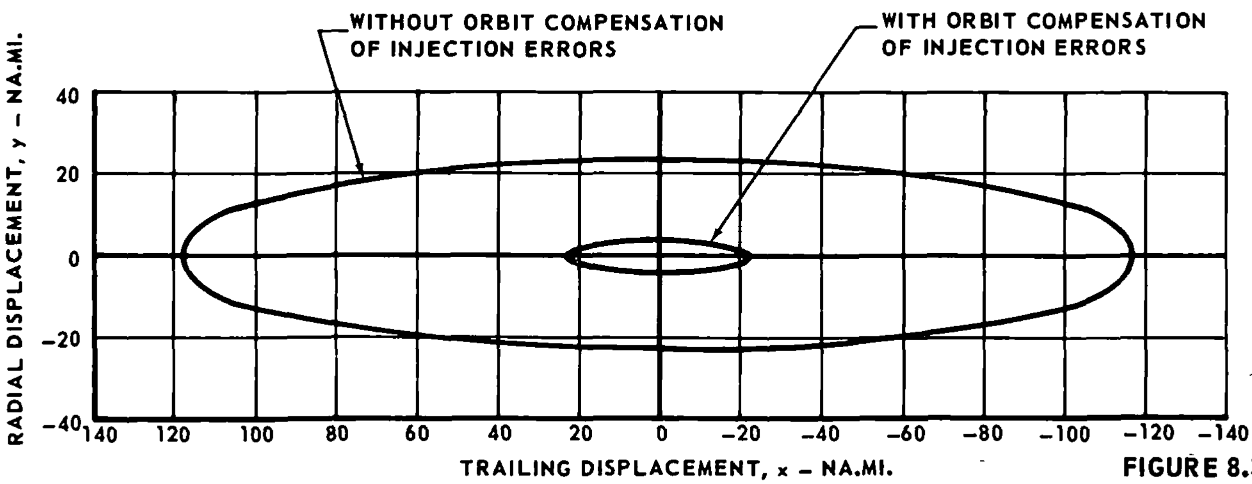


FIGURE 8.3-1

8.3.2 (Continued)

The ΔV required for rendezvous (and docking) from a nominal 87 to 250 na. mi. catch-up orbit into a 250 na. mi. circular orbit is reduced from 839 to 702 fps with 3σ probability. Measurement accuracies for MISTRAM, (Reference 8.3-4), which is expected to be in use during the MORL time period, would further reduce the 702 fps by a small percentage.

Similar analyses for the Ferry/Supply Spacecraft were not made due to unavailability of dispersions for the Saturn launch vehicles.

As shown in Figure 8.3-2, the launch time increment, (the time when the launch site is in the launch window and when the station-to-site angular phasing is proper for launch) is reduced by about two minutes, out of a minimum window of about 37 minutes at 250 na. mi., due to the allowance of a minimum of two orbits to make the in-orbit corrections. However, for operational launch vehicles this should not preclude use of the described method.

LAUNCH TIME INCREMENT AS A FUNCTION OF STATION ORBIT ALTITUDE

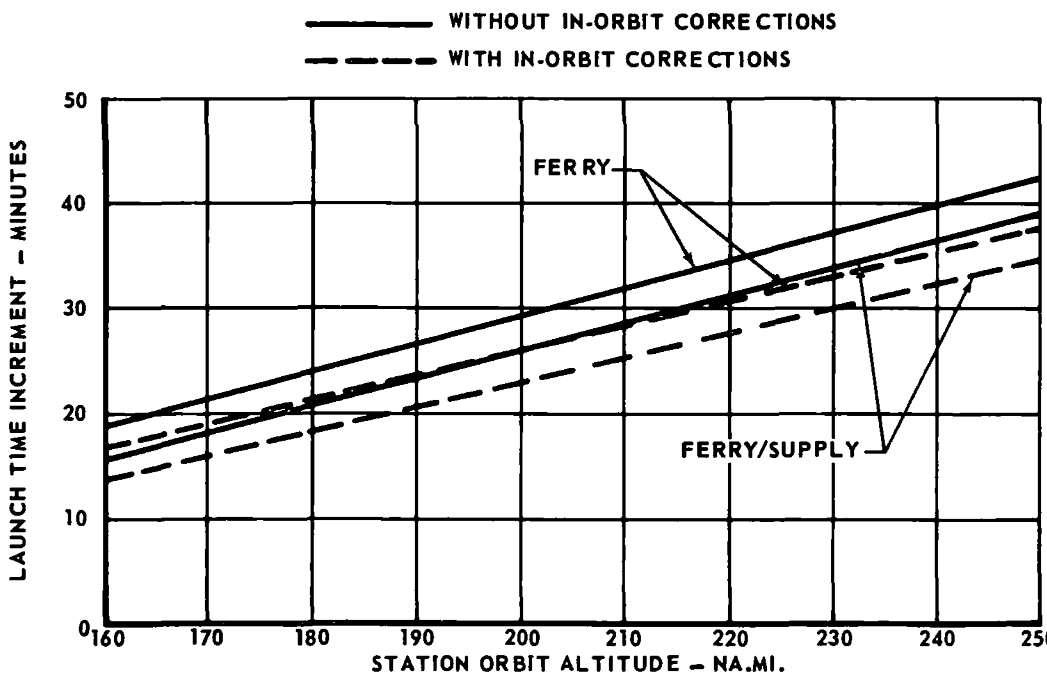
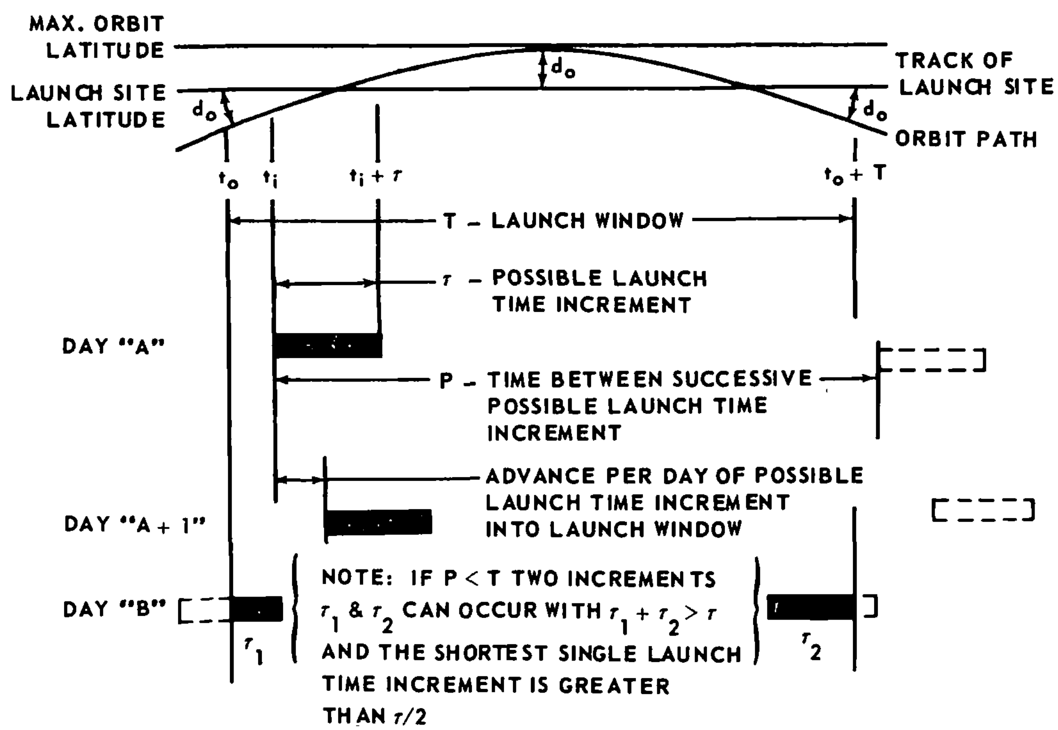


FIGURE 8.3-2

8.3.3 Reduction of Out-of-Plane at Injection - Two other methods of lowering rendezvous ΔV requirements involve reduction of the out-of-plane distance at injection from the 0.53° used for Gemini. Reducing the out-of-plane at injection primarily reduces the magnitude of the terminal rendezvous pulse. However, since the terminal rendezvous pulse is the vector sum of the out-of-plane component and in-plane error components, the net reduction in ΔV is less than calculated with errors neglected.

The first of these methods is to reduce the allowable out-of-plane at launch. This reduces the length of the launch window and can cause the allowable launch time increment for some days to be divided into two intervals. The relations between launch site latitude and the station orbit, and between the launch window and the possible launch time increments are graphically illustrated in Figure 8.3-3.

RELATIONS OF LAUNCH WINDOW AND LAUNCH TIME INCREMENT



LAUNCH WINDOW – TIME INCREMENT WHEN LAUNCH SITE IS WITHIN A GIVEN DISTANCE d_o OF THE ORBIT PLANE.

POSSIBLE LAUNCH TIME INCREMENT – TIME INCREMENT FROM WHEN INJECTION POINT OF FERRY LAGS STATION ENOUGH TO ALLOW ORBIT CORRECTIONS PRIOR TO AND FOR RENDEZVOUS WITHOUT LEADING THE STATION UNTIL THE STATION LEAD BUILDS UP TO WHERE THE FERRY CAN NO LONGER CATCH UP IN THE MAXIMUM ALLOWED TIME.

LAUNCH IS ALLOWED WHEN τ, OR DURING THE PART OF IT, FALLS WITHIN T.

FIGURE 8.3-3

8.3.3 (Continued)

The relationship between out-of-plane and launch window, for the case of station orbits selected to maximize ferry launch windows at Cape Canaveral, is shown in Figure 8.3-4.

EFFECT OF OUT-OF-PLANE ON LAUNCH WINDOW

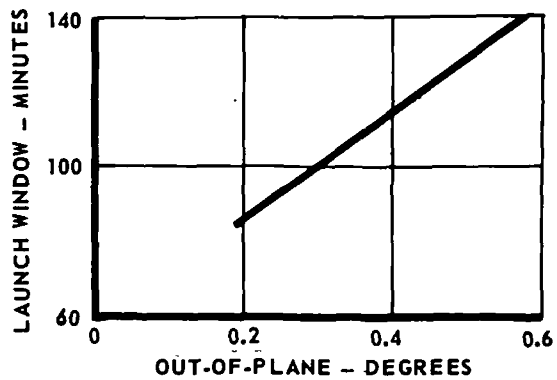


FIGURE 8.3-4

Daily launch time increments with 0.30 degrees out-of-plane are compared in Figure 8.3-5 with the minimum daily launch time increment currently available for Gemini. At 250 na. mi. reduction of the out of plane from the 0.53° of Gemini to 0.30° still provides a minimum continuous increment as large as for Gemini and results in 100 fps reduction in rendezvous ΔV.

The second method of reducing out-of-plane at injection is the use of launch vehicle yaw steering. During the study, insufficient information was available to permit a definitive analysis of this mode of launch. Indications are, however, that ΔV requirements can be reduced. Further investigation of this method appears desirable. (Section 14).

8.3.3 (Continued)

DAILY LAUNCH TIME INCREMENTS

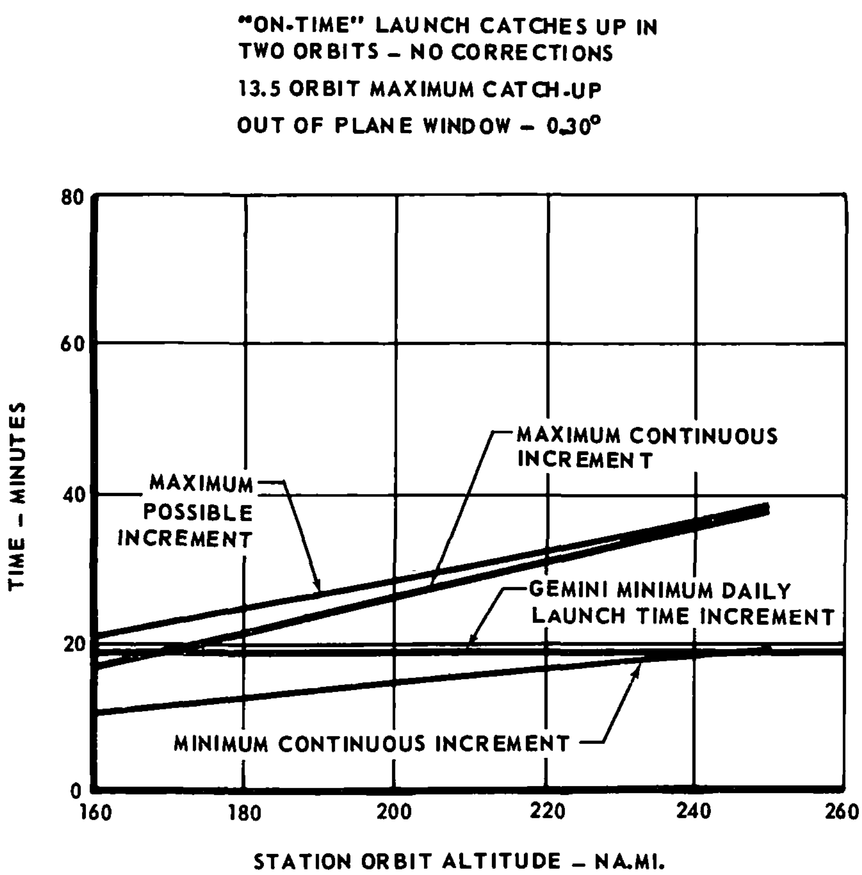


FIGURE 8.3-5

8.4 Separation Mechanics Analysis

8.4.1 Normal Separation - Separation of Ferry Spacecraft normally will occur when the MORL is not rotating. The spacecraft, at time of departure, is either nose or aft docked depending upon the configuration. Two solid rockets with an incremental velocity capability of approximately 5 fps each provide separation.

Since the thrust line of an individual rocket does not pass through the spacecraft center of gravity, rotational moments result if one rocket fails to fire. Analyses were made to determine separation dynamics for this case for the nose docked Ferry, which has the longest moment arm. Figure 8.4-1 shows that satisfactory separation from the docking cone occurs.

FERRY SEPARATION FROM DOCKING CONE WITH ONE SEPARATION ROCKET FAILURE

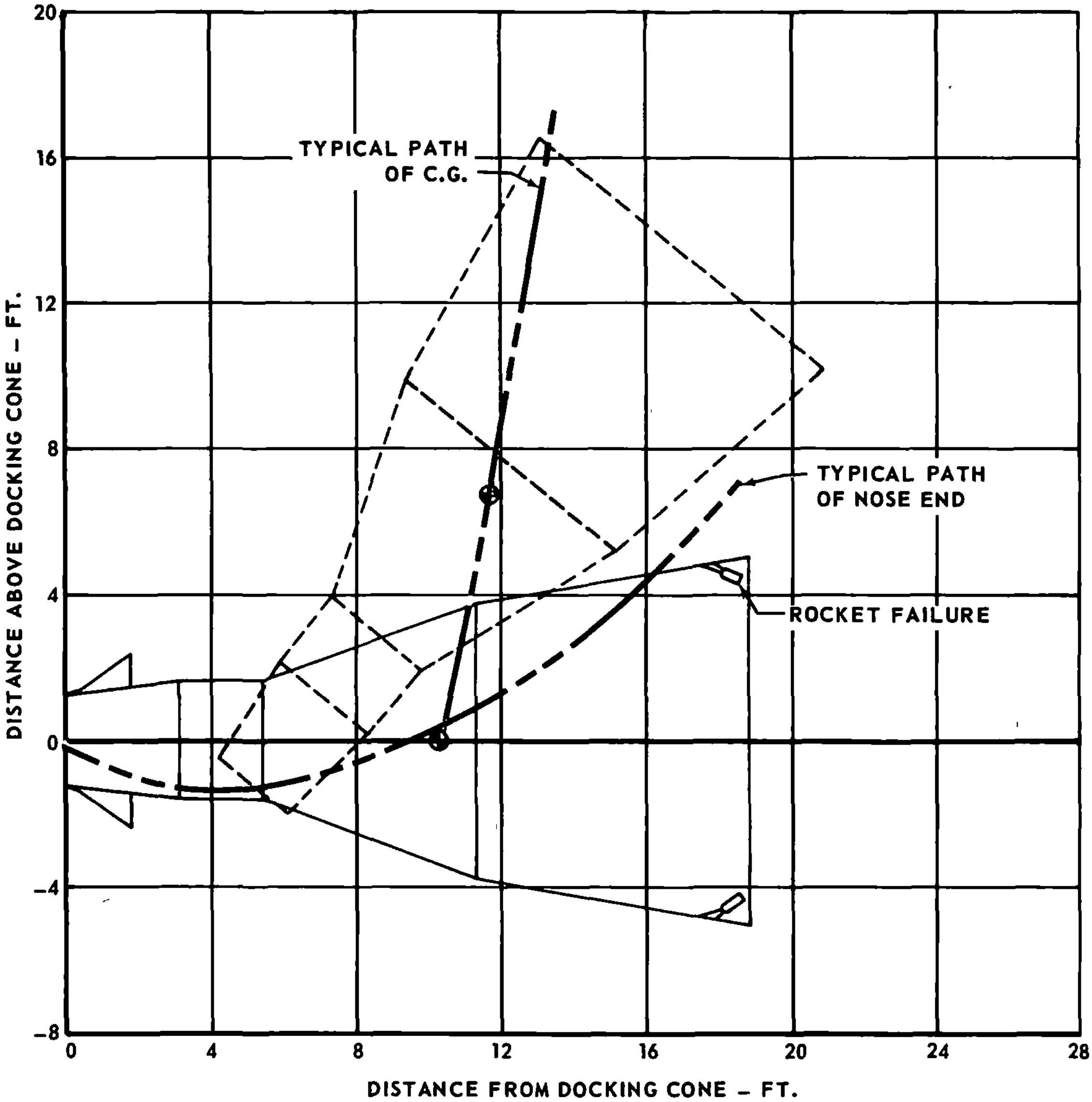


FIGURE 8.4-1

8.4.1 (Continued)

Following separation, residual tumbling rates are damped by the Re-entry Control System. (Section 7.1.3).

8.4.2 Emergency Separation - Two methods of separation from a station rotating with a counterbody were analyzed. The first method is to separate while rotating using the separation rockets. The ratio of separation velocity to rim velocity as a function of station and ferry mass parameters that will allow separation without counterbody contact is shown in Figure 8.4-2. The same ratio as a function of station physical parameters that will allow separation without station recontact is shown in Figure 8.4-3. Points representative of expected configurations are shown on each figure. Contact with the counterbody will not occur even if both rockets fail to fire. Recontact with the station may occur, however, if one rocket fails.

The second, and preferred, method of emergency separation is to cut the cable at the station thus separating the ferry-station combination from the counterbody. The Ferry is then separated from the station in the normal manner. This method involves the possibility of a collision between the counterbody and the ferry-station combination after 180° of orbit travel. The expected altitude difference after 180° orbit travel versus the angle between the cable and the orbit plane at the time of cable cutting is shown in Figure 8.4-4. Recontact will not occur unless the angle between the cable and the orbit plane, at the time of cutting, is small. Since the station is rotating at about 4.43 revolutions per minute, the cable cutting can be timed to avoid this region of potential recontact.

FERRY SEPARATION VELOCITY FROM STATION TO AVOID COUNTERBODY CONTACT

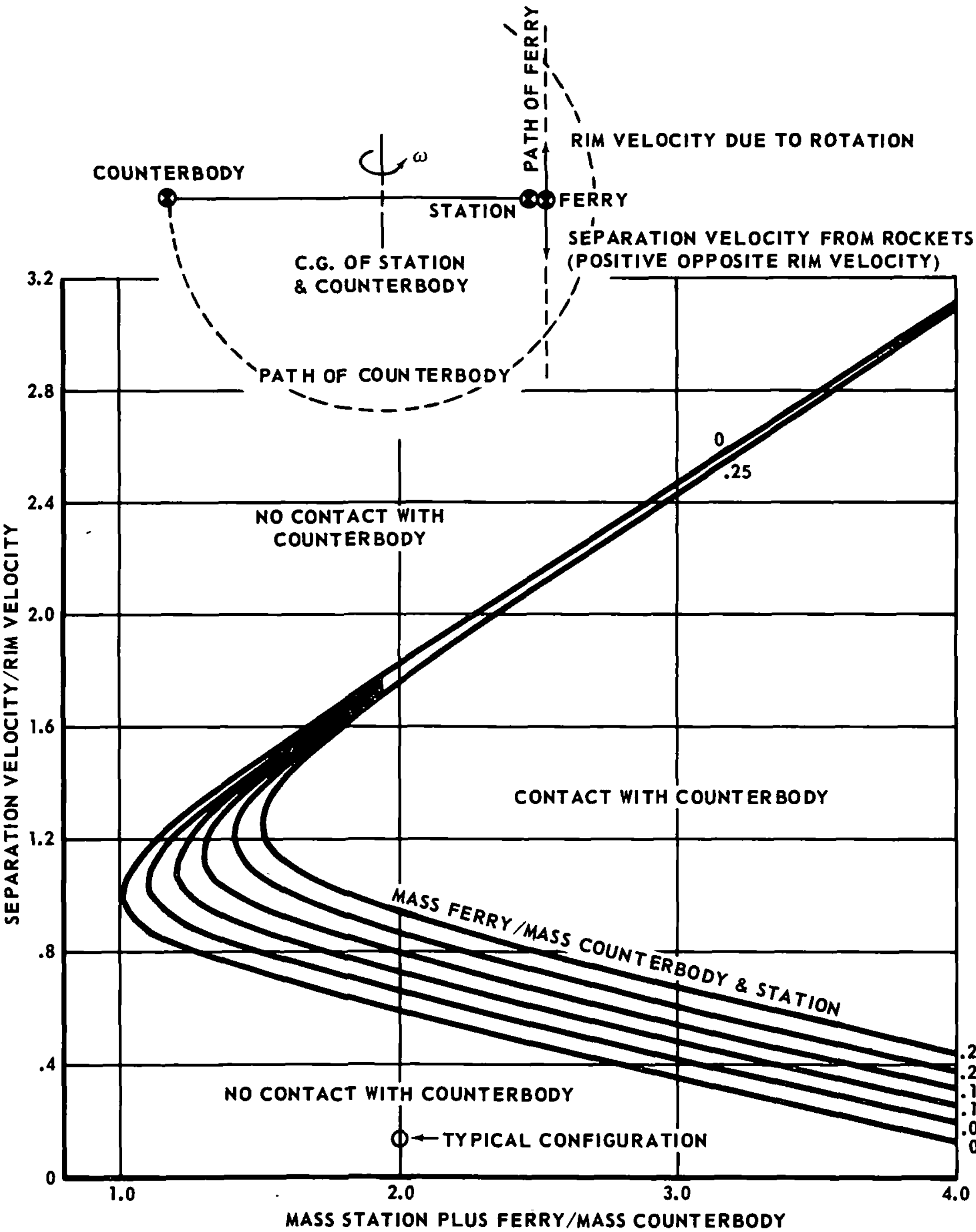


FIGURE 8.4-2

FERRY SEPARATION VELOCITY FROM STATION TO AVOID STATION CONTACT

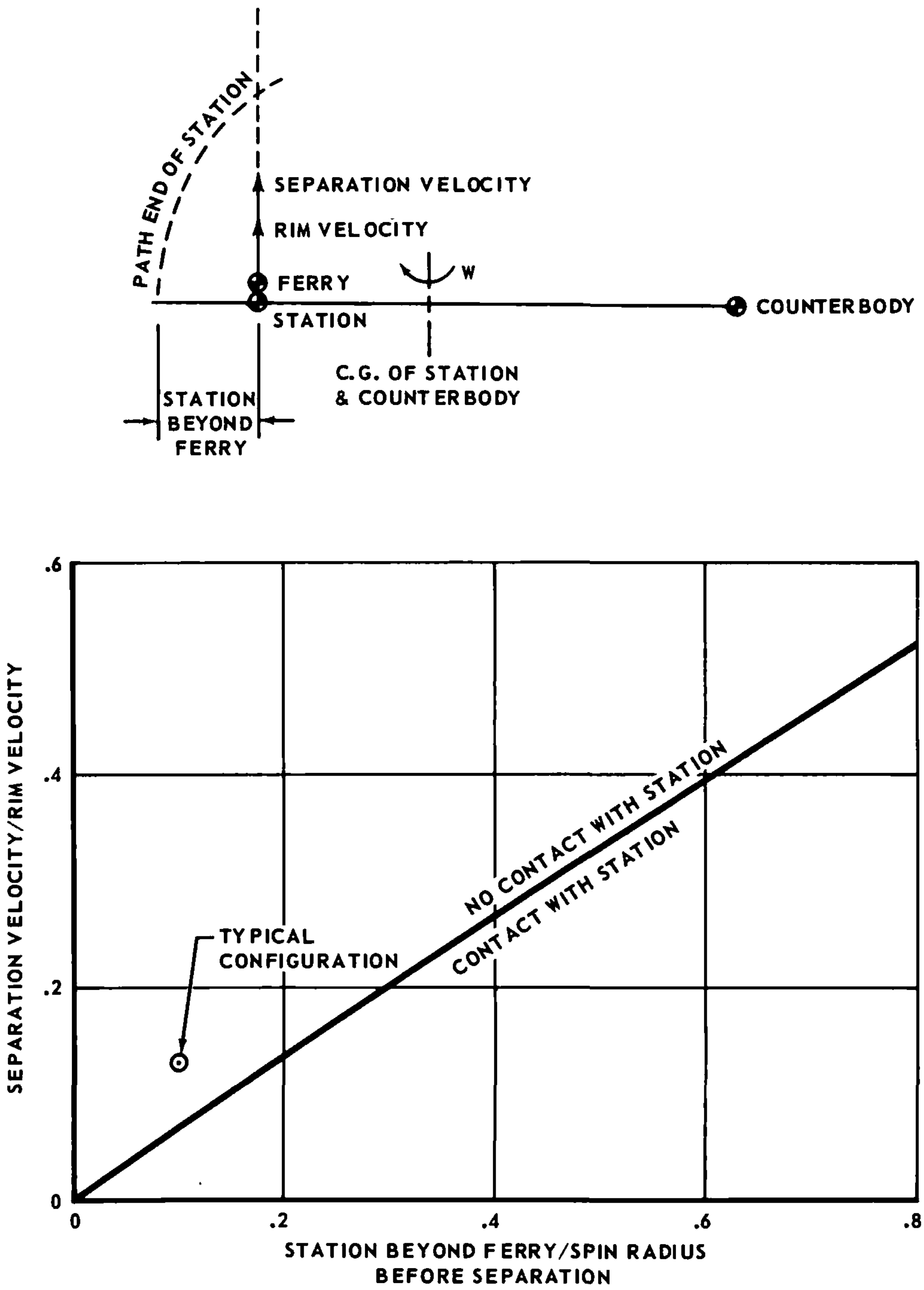
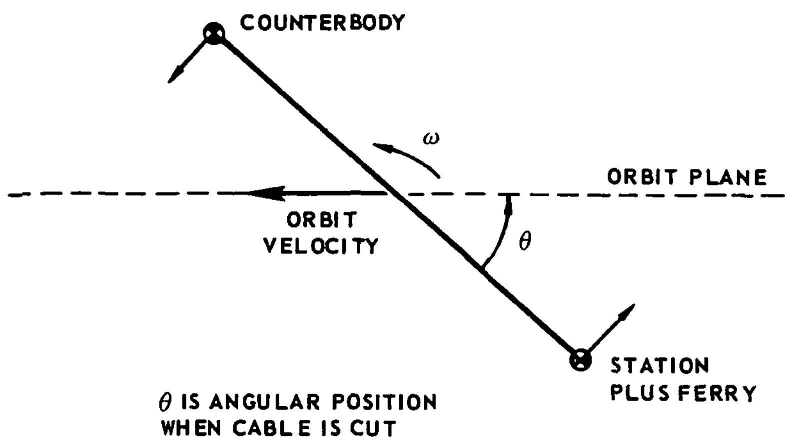


FIGURE 8.4-3

ALTITUDE DIFFERENCE BETWEEN FERRY AND COUNTERBODY AFTER 180° ORBIT TRAVEL vs ANGULAR POSITION WHEN CABLE IS CUT

70 FT./SEC. RIM VELOCITY
250 NA.MI. CIRCULAR ORBIT



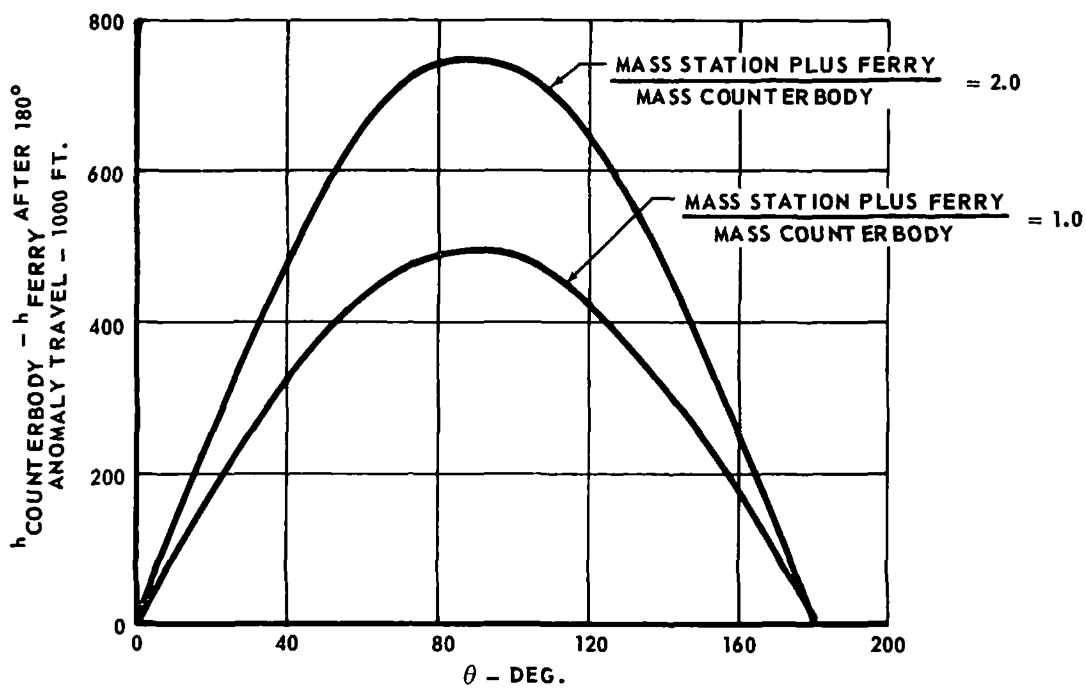


FIGURE 8.4–4

8.5 Retrograde and Re-entry Analysis

8.5.1 Retrograde Velocity Increments - The minimum retrograde velocity needed to re-enter from any orbit is determined by the skip boundary and the heat shield total heat capability.

The re-entry conditions at 400,000 feet altitude following retrograde from circular orbits, using four, five or six Gemini rockets, are shown in Figure 8.5-1. Re-entry conditions after retrograde from perigee and apogee, respectively, of the elliptical catch-up orbits are presented in Figures 8.5-2 and 8.5-3. To avoid the skip boundary, set from re-entry control considerations, five retrorockets are needed for a 250 na. mi. circular orbit and the associated catch-up orbit.

RETROGRADE FROM CIRCULAR ORBIT

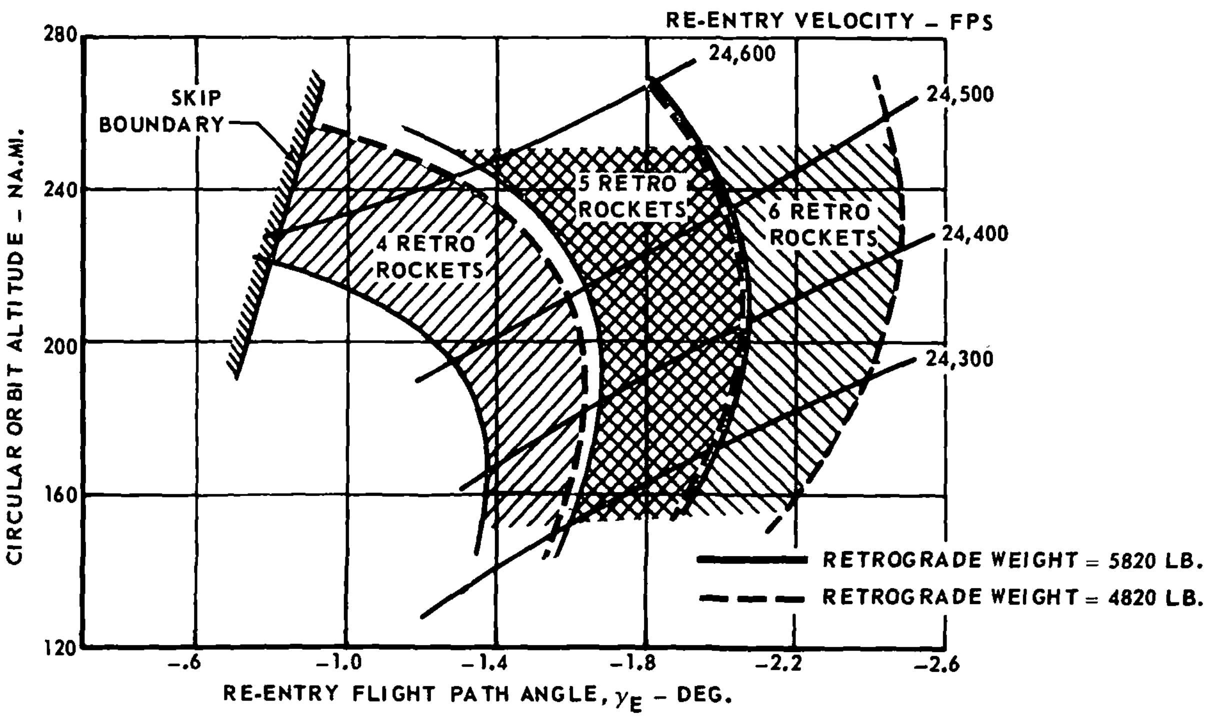


FIGURE 8.5-1

RETROGRADE FROM PERIGEE – ELLIPTICAL CATCH-UP ORBIT
PERIGEE ALTITUDE = 87 NA. MI.

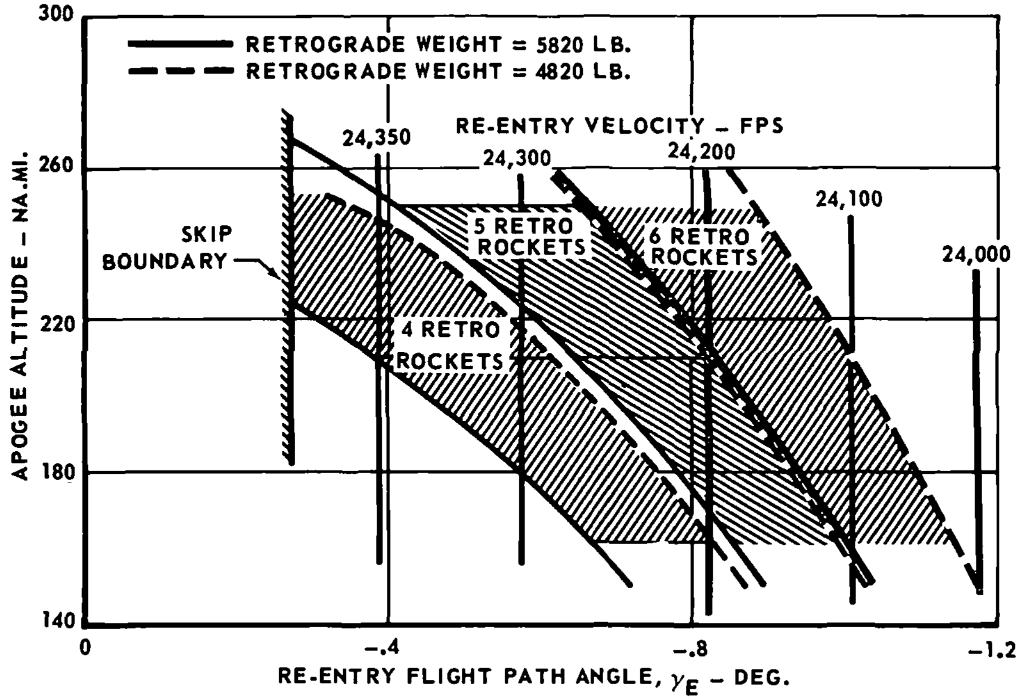


FIGURE 8.5–2

RETROGRADE FROM APOGEE – ELLIPTICAL CATCH-UP ORBIT
PERIGEE ALTITUDE = 87 NA.MI.

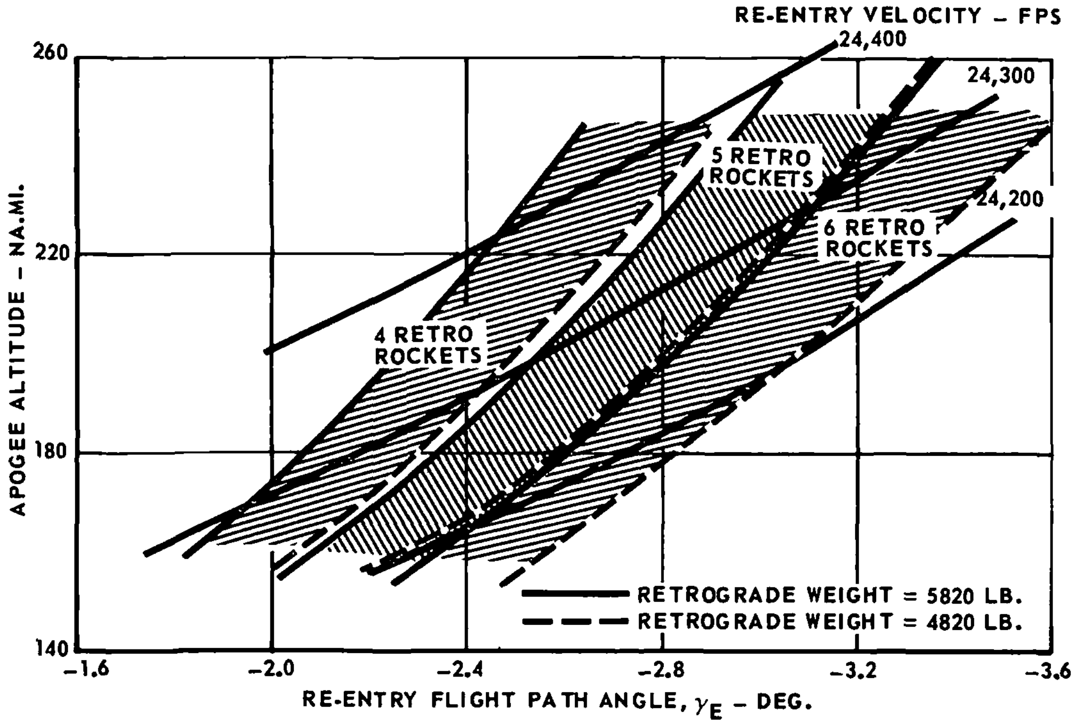


FIGURE 8.5–3

8.5.2 Re-entry Heating - The re-entry aerodynamic heating environment of the Ferry Spacecraft is essentially the same as that of Gemini. Minor differences occur due to re-entry from the higher orbital altitude of the MORL station. The heating is slightly more severe since, depending on the number of retrorockets used (Section 8.5.1), re-entry speeds are up to about 300 feet per second greater than Gemini; and flight path angles range from about -0.4° to -3.2° compared to about -0.67° to -2.1° for Gemini. This close similarity between re-entry conditions justifies the use of extrapolations of Gemini heating data.

Three methods of heat protection are used for the re-entry module: (1) an ablative heat shield, (2) radiation cooled conical afterbody (cabin) shingles of Rene' 41 and (3) beryllium heat sink shingles on the Re-entry Control and Rendezvous and Recovery Sections.

Ablative Heat Shield - The heat shield ablative material is an elastomeric silicone (DC325 retained in fiberglass honeycomb) which, in effect, acts as a high temperature insulation to protect the spacecraft structure from excessive temperatures. The thickness of the shield is designed to limit the maximum ablative material fiberglass honeycomb structure bondline temperatures to 500°F on the lower section of the heat shield at parachute touchdown, and 600°F on other areas. A 15% safety factor on heating rate is applied.

High ablative heat shield temperatures are produced by shallow re-entries. The bondline temperature of the present Gemini heat shield would exceed the 500°F limit for the following two re-entries: (1) retrograde from the 250 nautical mile circular orbit of the MORL using five retrorockets with L/D greater than 0.06 during flight in the atmosphere; (2) retrograde from perigee of the catch-up orbit (an "abort" condition) using five retrorockets with zero to maximum lift. By changing the taper of the ablative material from the present variation of 0.85 inch on the leeward side - 1.0 inch on the windward side to 0.85 inch - 1.15 inches, at a weight increase of 18 pounds, the bondline temperature in the lower section

8.5.2 (Continued)

of the heat shield at parachute touchdown can be maintained below the 500°F limit for: (1) retrograde from the 250 nautical mile circular orbit of the MORL station using zero to maximum lift; and (2) retrograde from perigee of the catch-up orbit using zero lift. The change in thickness is readily accomplished in the final machining operation of the ablative heat shield manufacturing procedure. The effect of the 18 lb. of ablative material on the heat shield re-entry boundary is shown in Figure 8.5-4. Bondline temperatures can also be maintained within design allowables by use of six Gemini retrorockets, or by restricting the amount of lift usable during re-entry.

RETROGRADE FROM CIRCULAR ORBITS
MAXIMUM LIFT RE-ENTRY - PARACHUTE WATER LANDING
RETROGRADE WEIGHT = 5800 LB.

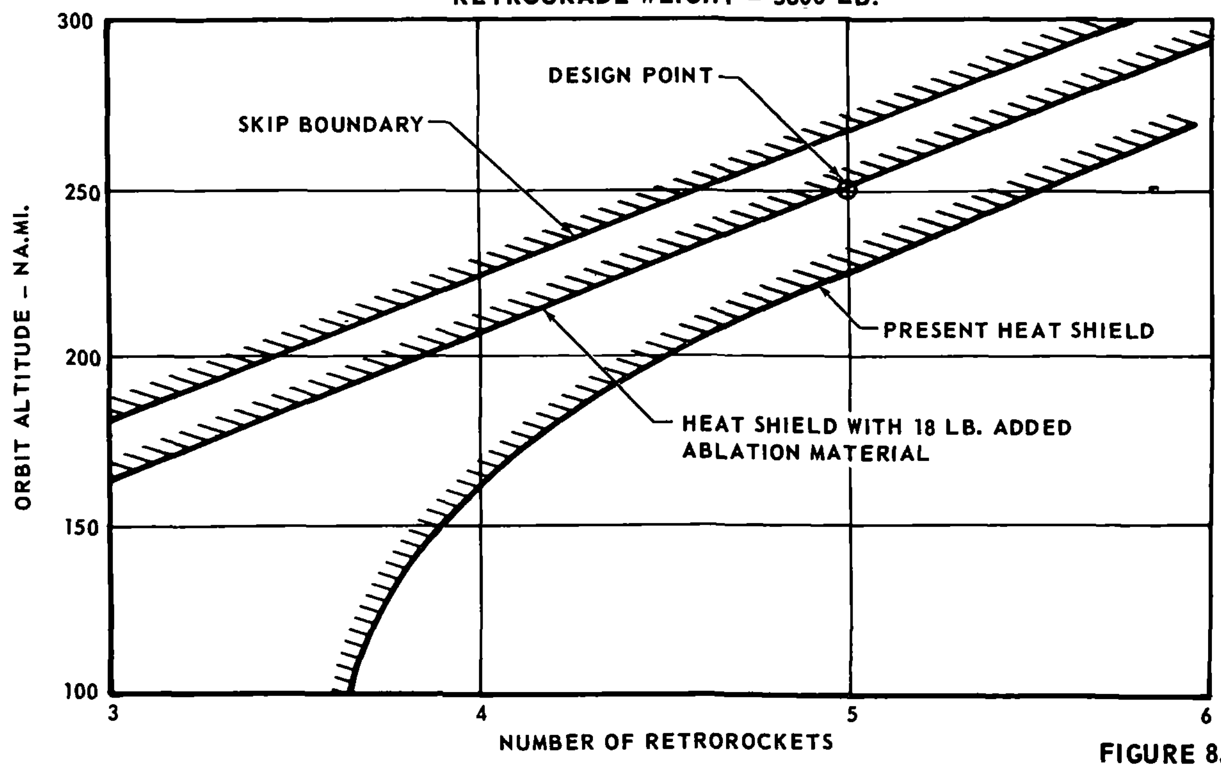


FIGURE 8.5-4

The aft docked spacecraft utilizes a hatch in the ablative heat shield for crew transfer. Considerable testing has been performed to determine whether or not such a hatch will affect the performance of the heat shield during re-entry

8.5.2 (Continued)

(Reference 8.5-1). Results of the test program (Reference 8.5-2) definitely indicate that a hatch in the ablative shield is feasible and that performance is not degraded.

Conical Afterbody - The conical afterbody is the same construction as Gemini and is covered with Rene' 41 radiation cooled shingles. Fibrous insulation is placed between the shingles and inner titanium alloy structure. Maximum temperatures occur on the bottom shingles (windward side at trim angle of attack) when the spacecraft rolls to obtain an effective zero-lift at steep re-entry flight path angles. The design allowable temperature for the Rene' 41 shingles is 1800°F (Section 7.3).

Retrograde from apogee of the catch-up orbit, using zero lift while in the atmosphere, results in the maximum afterbody temperature of 1755°F. The maximum temperature during re-entry from the 250 nautical circular orbit is 1620°F.

Rendezvous and Recovery and Re-entry Control System Sections - The bottom beryllium shingles require a 0.10 inch increase in thickness to maintain the temperature below the design allowable of 1600°F during a zero lift re-entry from perigee of the catch-up orbit. Other re-entries result in lower temperatures.

8.5.3 Retrorocket/Ablative Material/Operational Restriction Tradeoffs - Pertinent data from Sections 8.5.1 and 8.5.2 are summarized in Figure 8.5-4 which shows several possibilities for providing for re-entry from 250 na. mi. To avoid the skip boundary, about 4.7 retrorockets, or five, since they are discrete units, are needed. Providing a redundant rocket raises this number to six.

To maintain heat shield bondline temperatures with design allowables, four alternatives exist: (1) restrict the amount of lift usable during re-entry; (2) add 18 pounds of ablative material to the heat shield; (3) add 45 pounds of ablative material to the heat shield; or (4) add yet another retrorocket. Each of these techniques also affects temperatures encountered during

8.5.3 (Continued)

re-entry from elliptical catch-up orbits as shown in Figure 8.5-5. The features of the four alternatives are summarized in Table 8.5-1. Alternative (2) is the selected method.

HEAT PROTECTION LIMITS
5 RETROROCKETS

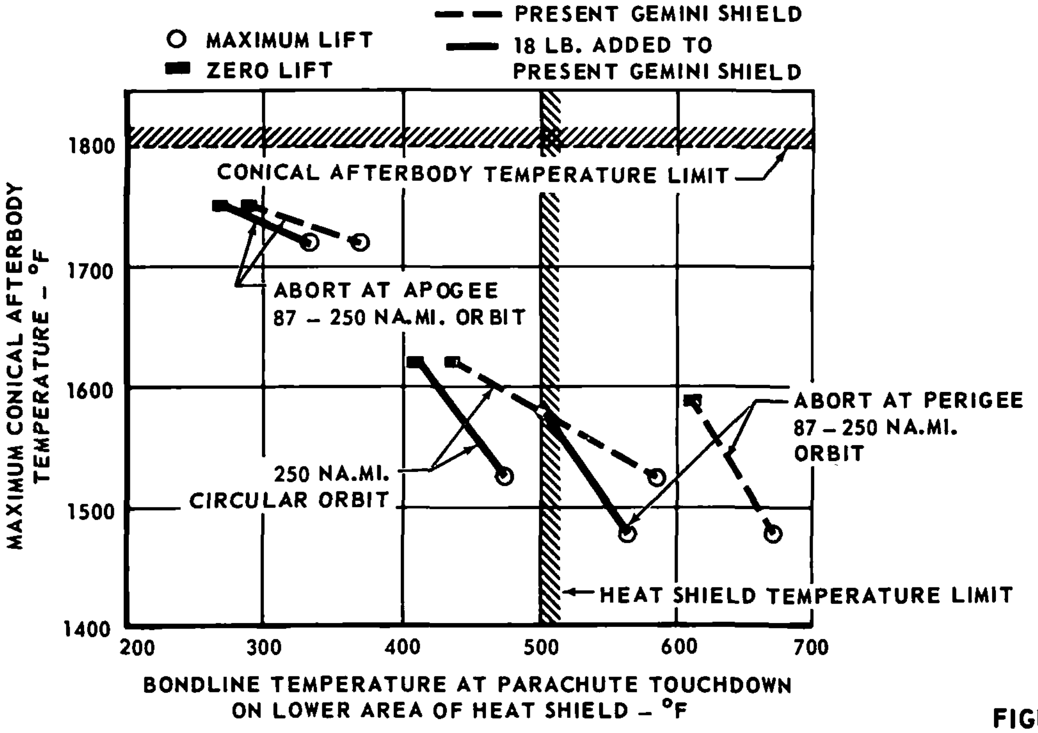


FIGURE 8.5-5

Alternative (4) was not selected since the zero-lift restriction only applies to an abort condition.

8.5.4 Method Operation of Retro-Rockets - The preferred method of operation is to ripple fire five retrorockets and to fire the sixth manually only if a failure occurs. In this way, the nominal touchdown will remain the same whether or not there is a failure. If the nominal mode were to fire all six, and a failure occurred, overshoot would result. Failure detection is accomplished by the astronaut by counting the pulses during ripple fire.

8.6 Orbit Maintenance Requirements - Orbit maintenance for the MORL is accomplished with propulsion systems installed in the supply spacecraft. Calculated orbit decay based on assumed MORL drag characteristics and the ARDC 1959 atmosphere is presented in Figure 8.6-1. Minimum to maximum solar activity is known to cause the density to vary by at least a factor of 3 and will thus affect orbit decay by a like amount. The next period of maximum solar activity (most dense atmosphere) should occur about 1968-1969. The

TABLE 8.5-1

RETROGRADE ABLATIVE MATERIAL TRADEOFFS
FOR STATION AT 250 NA.MI. ALTITUDE

CONFIGURATION NUMBER OF RETROROCKETS HEAT SHIELD THICKNESS	WEIGHT ADDED LB.	SPACE-CRAFT RE-ENTERS	ABLATIVE MATERIAL ADEQUATE			COMMENTS
			CIRCULAR ORBIT	PERIGEE OF CATCH-UP ORBIT	APOGEE OF CATCH-UP ORBIT	
GEMINI: 3 OF 4 ROCKETS .85–1.00 IN. ABLATIVE MATERIAL	—	NO	—	—	—	INADEQUATE
1. 5 OF 6 ROCKETS .85–1.00 IN. ABLATIVE MATERIAL	165	YES	ZERO LIFT ONLY	NO	YES	LIMITATIONS TOO GREAT.
2. 5 OF 6 ROCKETS .85–1.15 IN. ABLATIVE MATERIAL	183	YES	YES	ZERO LIFT ONLY	YES	LIMITATION IS MINOR
3. 5 OF 6 ROCKETS .85 TO 1.28 ABLATIVE MATERIAL	210	YES	YES	YES	YES	NO RE-ENTRY LIMITS (HEAVIER THAN 2)
4. 6 OF 7 ROCKETS .85–1.00 IN. ABLATIVE MATERIAL	248	YES	YES	YES	YES	NO RE-ENTRY LIMITS (HEAVIER THAN 2)

RECOMMENDED DESIGN

NOTE: ONE REDUNDANT ROCKET CARRIED FOR RELIABILITY IN ALL CONFIGURATIONS.

8.6 (Continued)

CIRCULAR ORBIT DECAY

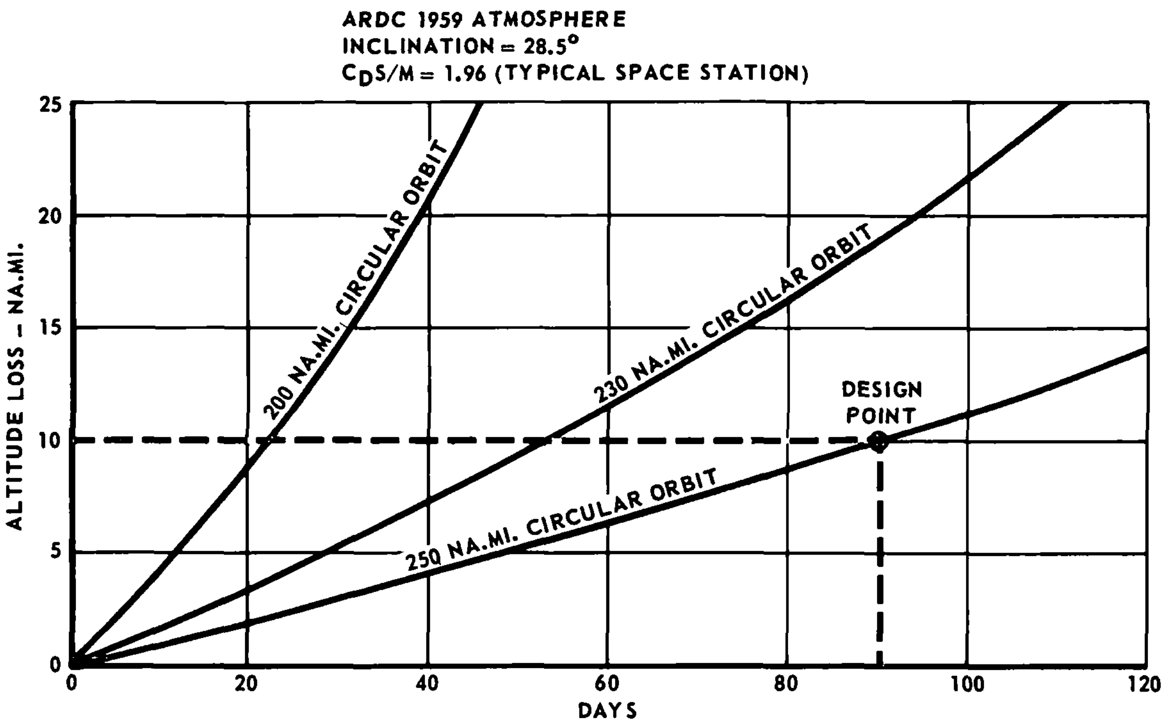


FIGURE 8.6-1

ARDC 1959 atmosphere was used since it gives a reasonable approximation for the 1968-1969 time period.

Velocity increments (ΔV) required to restore the decayed orbit to the original altitude were computed assuming Hohmann transfers, and are presented in Figure 8.6-2. A total ΔV of 137 fps per year with corrections approximately every 90 days will maintain the orbit within 10 na. mi. of a 250 na. mi. altitude. This value of ΔV, with slight variations to account for supply schedules, is used to calculate the amount of propellant per year to be supplied to the station.

8.7 <u>Saturn Escape Analysis</u>

8.7.1 <u>Saturn Abort Environment</u> - The Gemini launch escape system utilizes ejection seats for escape from the Gemini Launch Vehicle (GLV).

For Ferry/Supply Spacecraft, Saturn launch vehicle explosion overpressures and fireball heating will be much more severe than GLV, due both to greater pro-

ORBIT MAINTENANCE REQUIREMENTS

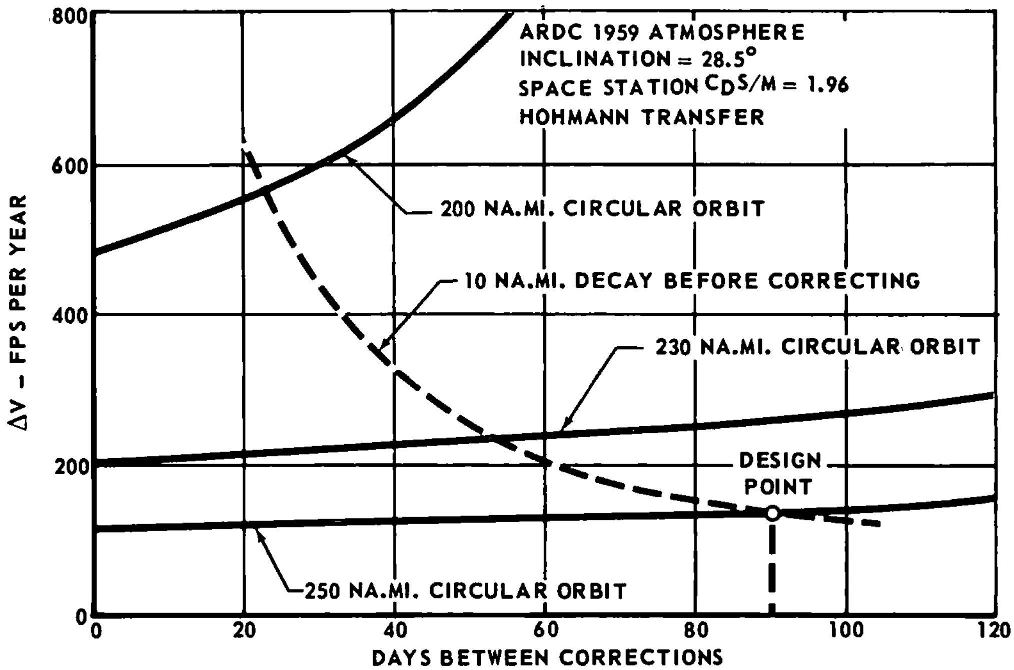


FIGURE 8.6-2

pellant quantity and the higher equivalent explosive yield. The blast wave overpressure pattern and maximum fireball radius resulting from a Saturn IB explosion compared with Gemini ejection seat trajectories is shown in Figure 8.7-1. The overpressures are calculated from data received informally from NASA on the explosive effect of liquid oxygen, liquid hydrogen and RP-1 mixtures and assume an equivalent explosive yield of .6 pounds of TNT per pound of total propellants aboard. The maximum fireball radius was estimated from the data of Reference 8.7-1. It was assumed, for both calculations, that both stages of Saturn mix and explode. Seat ejection is feasible for escape from the high overpressure region with about 2.2 seconds warning of an impending explosion. However, the seats do not have sufficient range to escape the region of excessive heating from the fireball, nor is it practical to design an ejection seat with sufficient range to provide escape from the fireball.

SATURN IB EXPLOSION OVERPRESSURE AND FIREBALL CHARACTERISTICS

▲ INITIATE PARACHUTE DEPLOYMENT – T = 3.5 SEC.

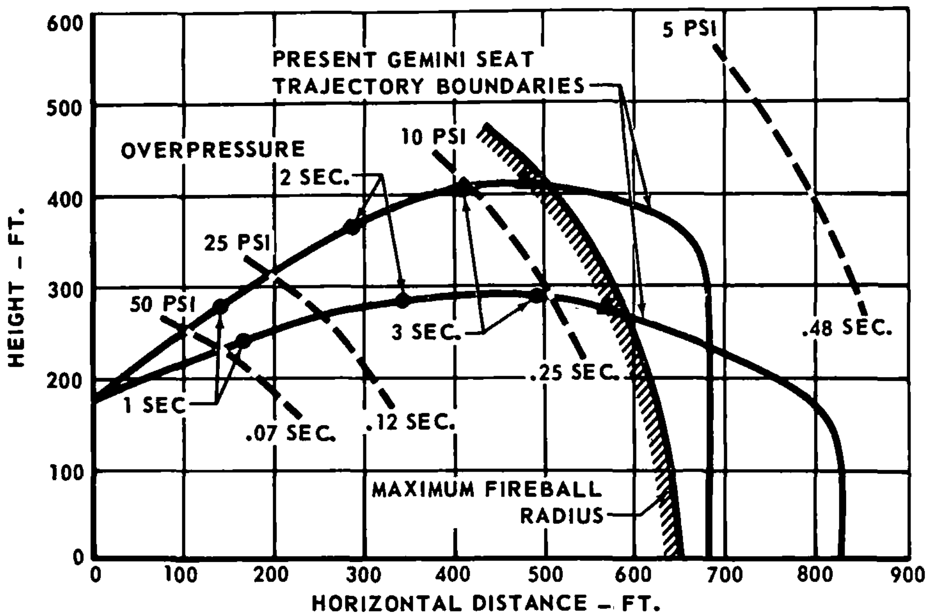


FIGURE 8.7-1

8.7.2 <u>Saturn Launch Escape System</u> - To provide escape capability, two launch escape system designs were investigated. One system consists of a tower mounted solid propellant escape rocket, similar to that used on the Mercury spacecraft, while the other utilizes 8 Cherokee rockets, externally mounted on the adapter section, together with salvo-firing of the retrograde rockets. Fins mounted on the adapter section provide aerodynamic stability for the latter configuration. Both systems are designed for the following requirements:

A. Off-the-pad escape trajectory having a minimum apogee of 3000 feet with a minimum range at apogee of 2800 feet.

B. Sufficient stability to maintain trajectory control.

C. Adequate separation under all flight conditions to avoid blast damage in the event of launch vehicle explosion.

D. Landing accomplished either by deploying a parachute or by seat ejection.

8.7.2 (Continued)

E. Emergency crew acceleration tolerances not to be exceeded.

The resultant escape system characteristics are summarized in Table 8.7-1.

TABLE 8.7-1
ESCAPE SYSTEM CHARACTERISTICS

	TOWER ESCAPE SYSTEM	FINNED ADAPTER ESCAPE SYSTEM
THRUST (RESULTANT), LB.	95,000	86,300
THRUST DURATION, SEC.	1.2	1.12 CHEROKEE 5.02 RETRO ROCKET
IMPULSE (RESULTANT), LB.-SEC.	156,000	168,200
MAX. LOAD FACTOR, g's	15.8	14.2

Off-the-pad escape trajectories for the finned adapter configuration are shown in Figure 8.7-2 with allowance for thrust eccentricity included, for the case of one rocket failure. Trajectories for the tower escape system are similar.

PAD ESCAPE CAPABILITY
FINNED ADAPTER ESCAPE SYSTEM

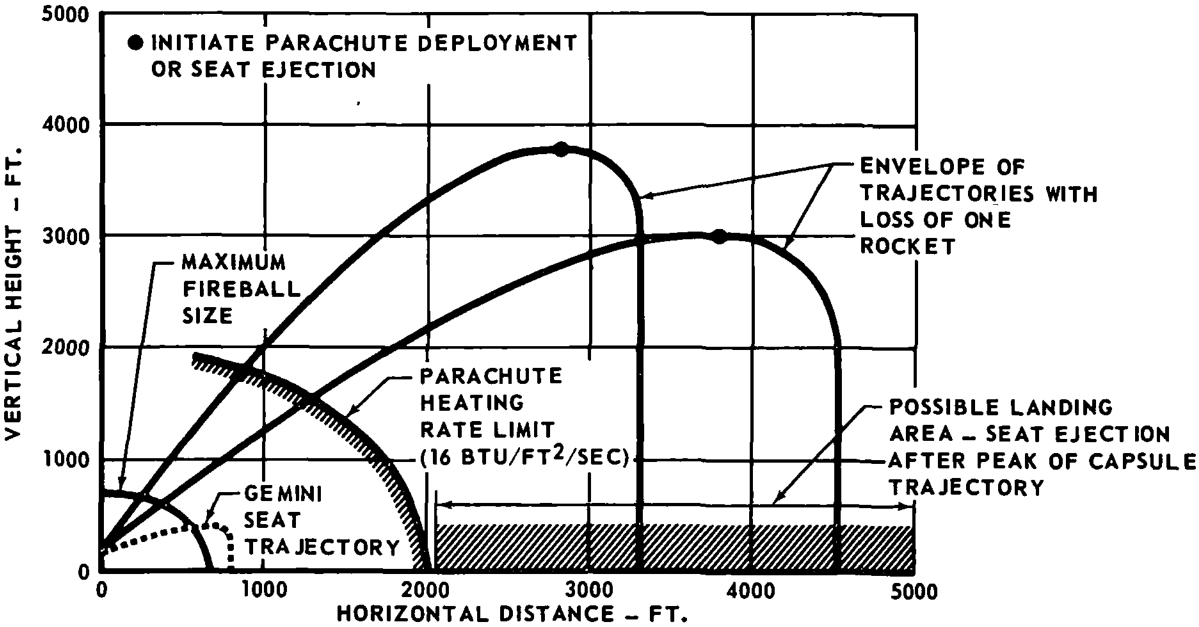


FIGURE 8.7-2

8.7.2 (Continued)

Maximum fireball size and parachute heating rate limit are also shown for comparison. The altitude attained is sufficient for parachute deployment in case of downward ejection, and the lateral range is sufficient to avoid the heating if ejection in the direction of the fireball occurred. The desired lateral range is attained, for the finned adapter configuration, by asymmetrically mounting four of the Cherokee rockets to provide a lateral thrust component. In addition, the thrust vectors of these four rockets are directed slightly above the escape configuration center of gravity to provide a pitching moment. For the tower escape system, eccentricity of the thrust vector with respect to the center of gravity is used.

Estimated subsonic and supersonic static stability characteristics of both escape configurations are shown in Figure 8.7-3. The tower configuration stability is based on Mercury spacecraft data while the finned adapter configuration is based on studies previously conducted of a similar configuration for Project Mercury.

8.8 Radiation Hazard

8.8.1 Environment - The radiation environment used in evaluation of the hazard associated with ferry missions is shown in Figure 8.8-1. The flux of natural protons and of natural and artificial electrons is presented as a function of orbit altitude for 30° inclination circular orbits.

Electrons - The electron flux results principally from beta particles which were emitted during the Starfish high altitude nuclear explosion of 9 July 1962 and subsequently trapped in the magnetic field of the earth. The highest intensity curve of Figure 8.8-1 is plotted from the results of a computer program which utilizes a flux model, constructed by W. Hess (Reference 8.8-1), based on the measurements obtained with satellites about one week after the explosion. The curve for July 1967 is an estimate of the environment assuming that there will

ESCAPE CONFIGURATION STATIC STABILITY

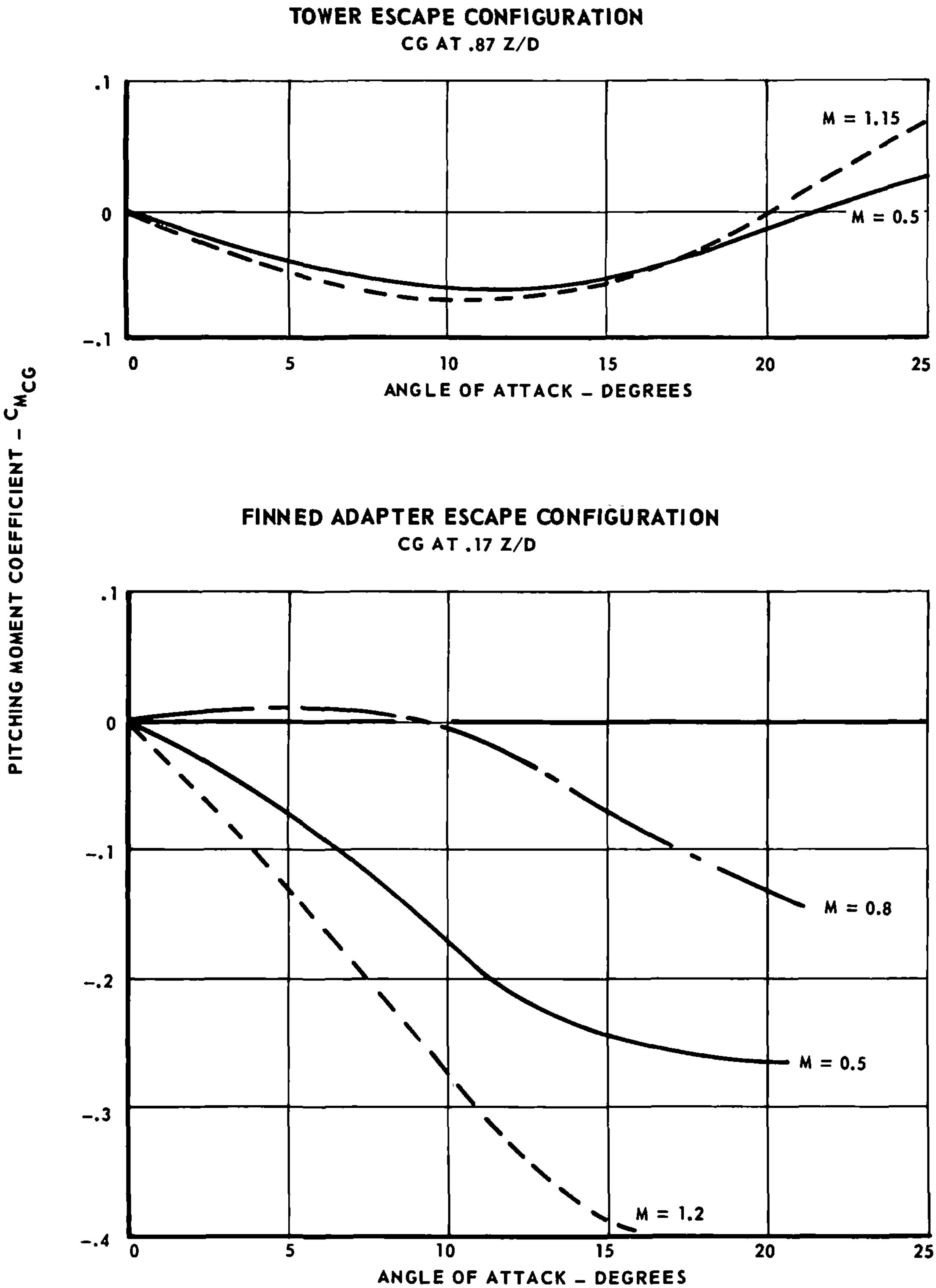


FIGURE 8.7–3

TRAPPED PARTICLE FLUX FOR 30° INCLINATION CIRCULAR ORBITS

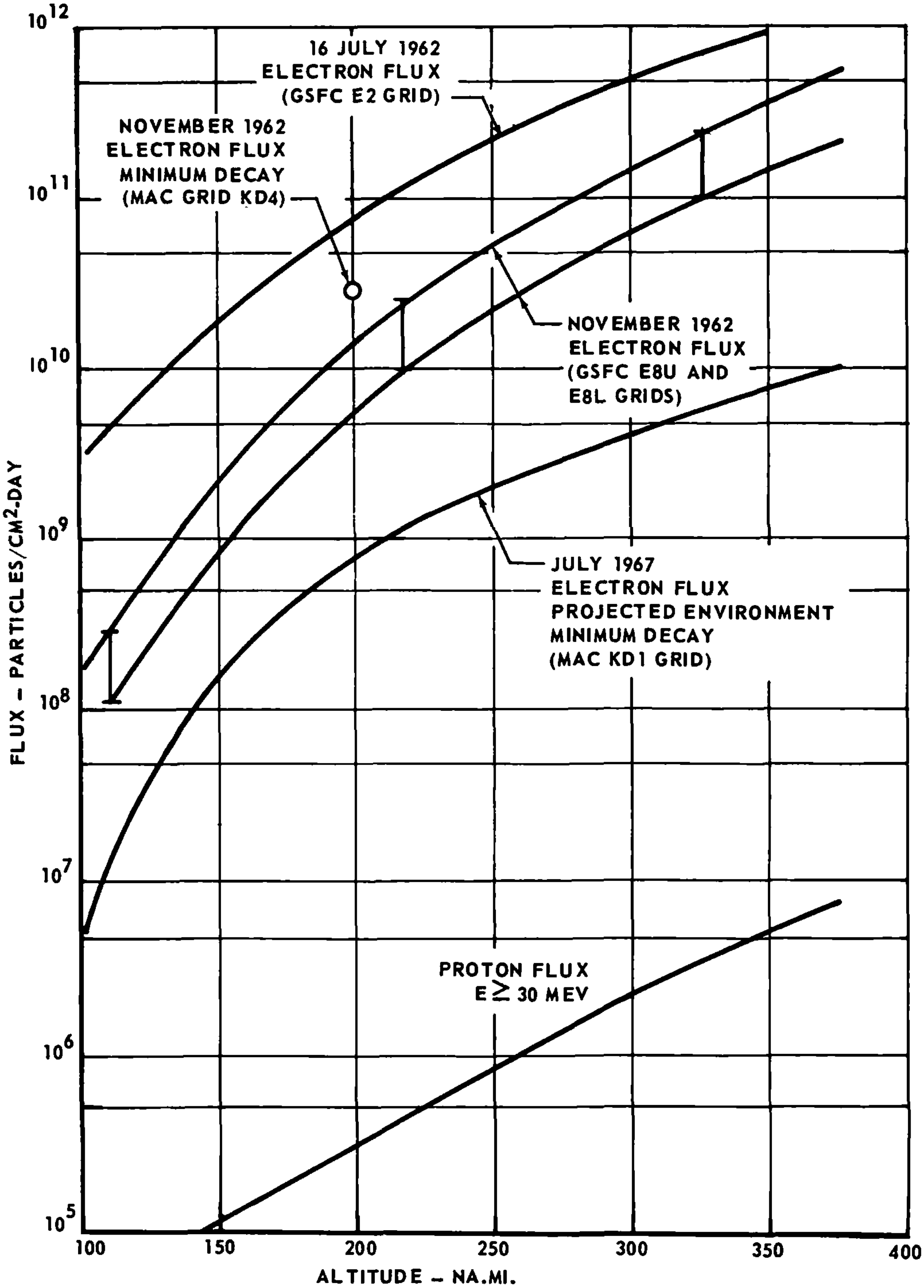


FIGURE 8.8-1

8.8.1 (Continued)

be no further high altitude nuclear testing of significance, and that decay of the trapped radiation will remain exponential with characteristic decay times as determined from the measured data through December 1962. Decay times measured at positions on the Geomagnetic Equator are used to provide a "minimum" decay model.

Due to the uncertainties associated with prediction of electron flux for the time period of MORL, two sets of dose predictions are included. One is based on the 16 July 1962 flux curves. Although not an upper limit to possible environments, it represents a demonstrated threat. The other is based on the predicted flux curves for 1967.

The electron spectra for 30°, 60° and 90° inclination circular orbits at various altitudes have been calculated by W. Hess for November 1962 (E8 grid - Reference 8.8-2). The E8 grid uses different energy spectra for different ranges of magnetic shell parameter, L, based on the available measured data. The results show that the spectra for circular orbits do not deviate greatly from a fission spectrum, particularly in the high energy range. A comparison between the fission spectrum (Reference 8.8-3) and the calculated spectrum for a typical case is shown in Figure 8.8-2. For simplicity the beta fission spectrum was used in analysis.

Protons - The flux of trapped protons with energies greater than 30 Mev, shown in Figure 8.8-1, is from data obtained using a computer program developed at McDonnell with a proton flux grid based on available measured data. The program output provides the energy spectrum for the range 30 to 600 Mev. The results indicate an extremely hard spectrum for the altitude range of 160 to 250 na. mi.

ELECTRON SPECTRA

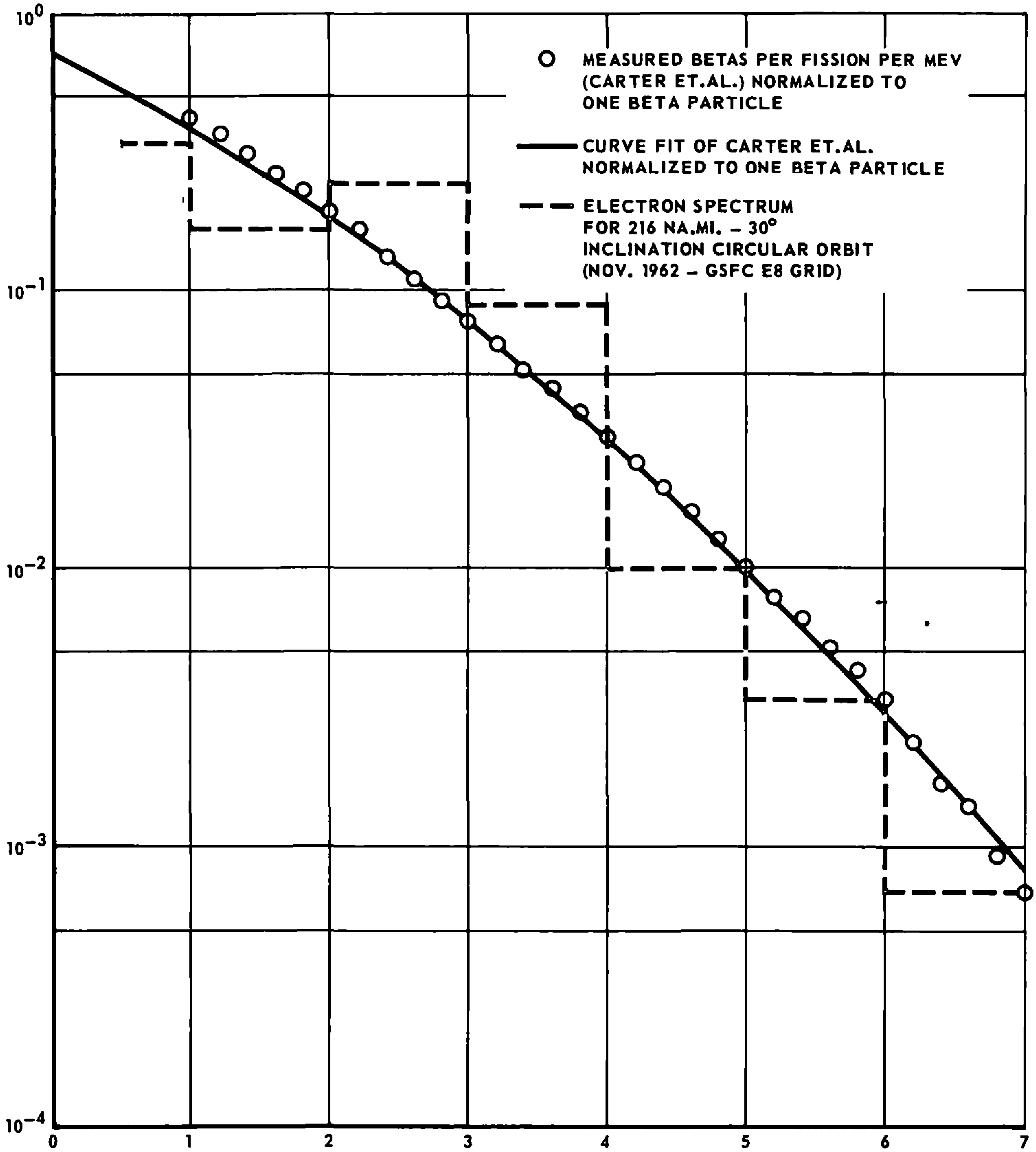


FIGURE 8.8-2

8.8.1 (Continued)

Time Variation - The variation with orbit number of the proton flux and electron dose rate for an unshielded omnidirectional unit detector is shown in Figure 8.8-3 for a 200 na. mi. circular orbit. The variation reflects the existence of anomalous regions of radiation intensity resulting from the unsymmetrical nature of the geomagnetic field. The flux to dose conversion factor of fission spectrum electrons is 4.28×10^{-8} rad/electron/cm^2.

The large variation in intensity with orbit number or time emphasizes the importance of scheduling, e.g., extravehicular activity, as an effective means of reducing radiation exposure for mission phases of a few hours duration. There is approximately a 16 hour period available per day during which the exposure is essentially minimum (Figure 8.8-4). During this 16 hour period, the spacecraft is in a low intensity region in which flux model is not applicable. The dosages shown in Figure 8.8-4 are the result of an extrapolation procedure contained within the computer program and are somewhat uncertain. In order to counteract this it is assumed that the lowest intensity contour of the electron flux model, 10^5 per cm^2 per hour, is applicable in the low intensity regions. Considering the protection provided by the pressure suit (assumed to be 1 lb./ft^2, attenuation factor of 0.28) this intensity corresponds to a dose rate of 9.2 rads/hr. when scaled up to 250 na. mi. orbit. For the July 1967 environment, inclusion of a minimum decay factor results in a dose rate of 0.72 rad/hr.

A corresponding procedure applied to the proton exposure results in the value 2.88×10^4 protons/cm^2-hr.

8.8.2 Shielding - (See also Appendix B). The inherent shielding afforded by the spacecraft structure and equipment was determined for a position on the astronaut's face. Using this point as reference, a 720 element equal solid angle grid was constructed.

UNSHIELDED ELECTRON DOSE RATE AND PROTON FLUX FOR A 200 NA.MI. CIRCULAR ORBIT

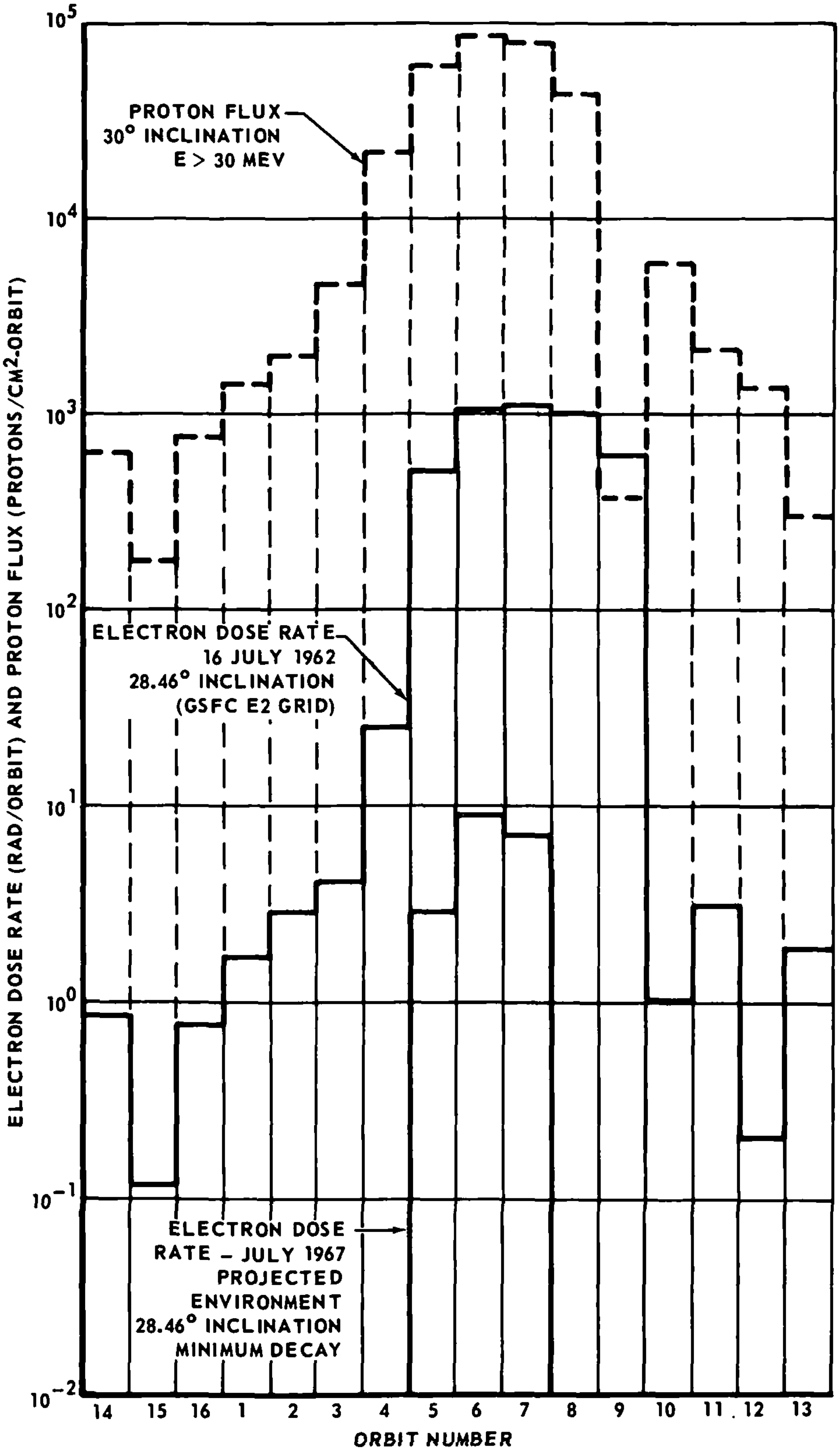


FIGURE 8.8-3

ELECTRON DOSE RATE – TIME VARIATION DURING TRANSFER

ENVIRONMENT BASED ON GSFC E2 GRID FOR 16 JULY 1962
200 NA.MI. CIRCULAR ORBIT – 28.46° INCLINATION
1 LB/FT² SHIELD THICKNESS

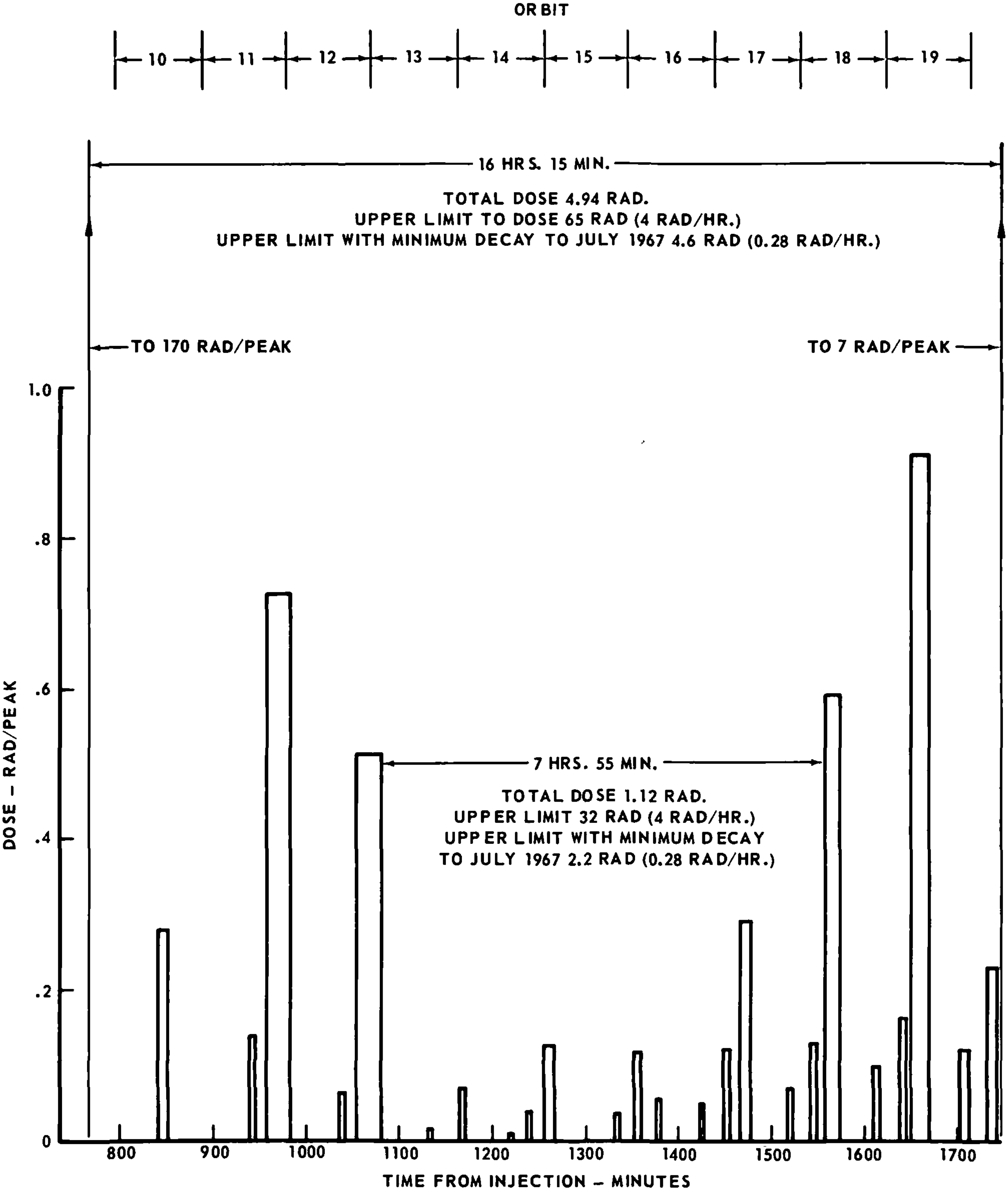


FIGURE 8.8-4

8.8.2 (Continued)

The summation of the material thicknesses (gms/cm^2) provided by the spacecraft structure and equipment and contained within any given solid angle element was obtained. The distribution shown in Figure 8.8-5 is the result of summing all the elementary solid angles with a given associated thickness.

GEMINI SHIELDING

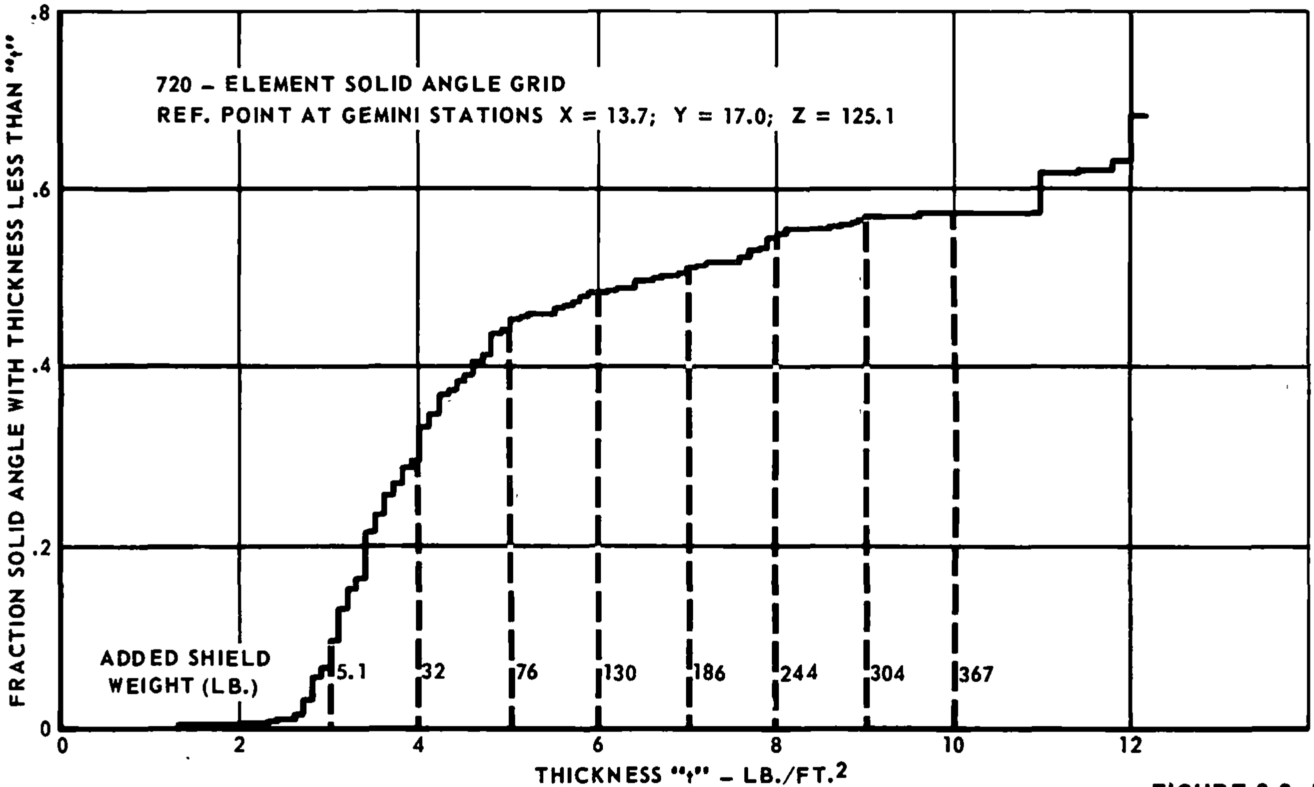


FIGURE 8.8-5

This distribution represents approximately 60% of the inherent spacecraft weight. The remaining 40% of the weight consists of equipment which is of minor importance in the determination of the over-all shielding effectiveness.

The change in the solid angle-thickness distribution with added shield weight is shown in Figure 8.8-5. Shield material is assumed to be added to sections of lesser thickness before any addition to sections of greater thickness. Therefore each distribution is characterized by a minimum shield thickness as well as an added shield weight. This procedure results in approximately

8.8.2 (Continued)

optimum addition of spacecraft shielding.

Experiment has shown that the absorption of beta ray spectra by matter, when expressed as a function of thickness in gms/cm^2 or areal density, is nearly independent of the atomic weight of the absorber and increases only slightly with increasing atomic number (References 8.8-4 and 8.8-5). The attenuation curve for aluminum (Figure 8.8-6) is therefore assumed to apply to the composite thickness - solid angle distributions described above. The effect of added shield weight on the calculated attenuation factor for the spacecraft is shown in Figure 8.8-7. The dose attenuation factor for electrons with inherent shielding only is found to be 9.2×10^{-3}.

The change with added shield weight of the relative importance of the bremsstrahlung dose in comparison to the penetrating electron dose is illustrated in Figure 8.8-7. The attenuation of bremsstrahlung radiation by aluminum, which is assumed to be representative of the actual materials, does not change significantly with thickness over the range considered (1.3 to about 4 gms/cm^2). Therefore, the bremsstrahlung dose is approximately independent of shield weight for the indicated range.

The attenuation curve for protons (Figure 8.8-8) is used in conjunction with the various solid angle thickness distributions to provided values of the dose per unit proton flux ($E \geq 30$ Mev) as a function of shield weight. The use of aluminum as a representative shield material is consistent with the method by which proton attenuation was determined i.e., ionization loss only. The results presented in Figure 8.8-9 indicate that the trapped protons are very difficult to attenuate compared to the electrons.

The values of surface dose per unit incident flux for the Ferry Spacecraft with inherent shielding are shown in Table 8.8-1.

DOSE RATE ATTENUATION FOR FISSION SPECTRUM ELECTRONS TRANSMITTED THROUGH A SLAB SHIELD

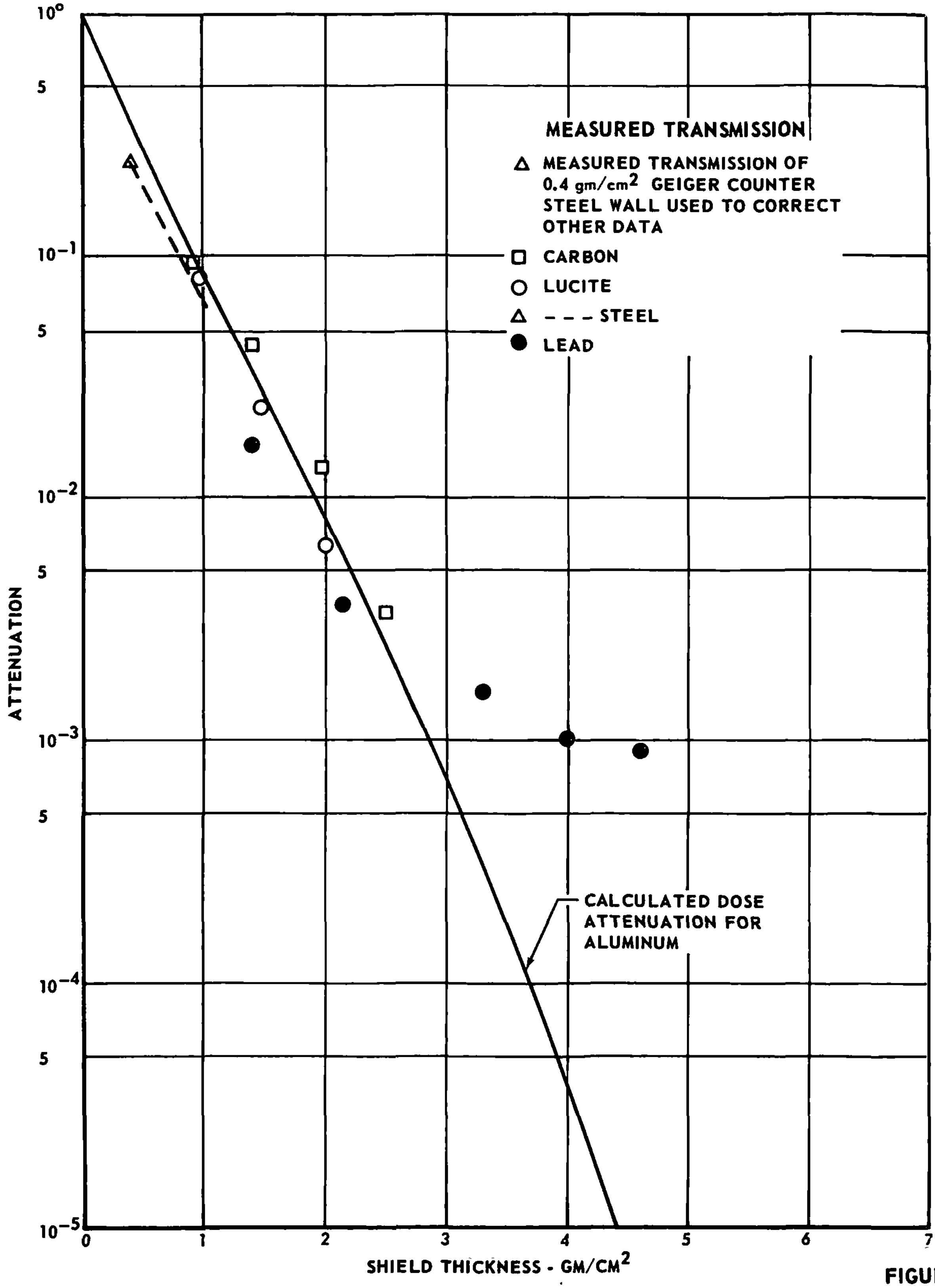


FIGURE 8.8–6

ELECTRON DOSE ATTENUATION FACTOR FOR FERRY SPACECRAFT

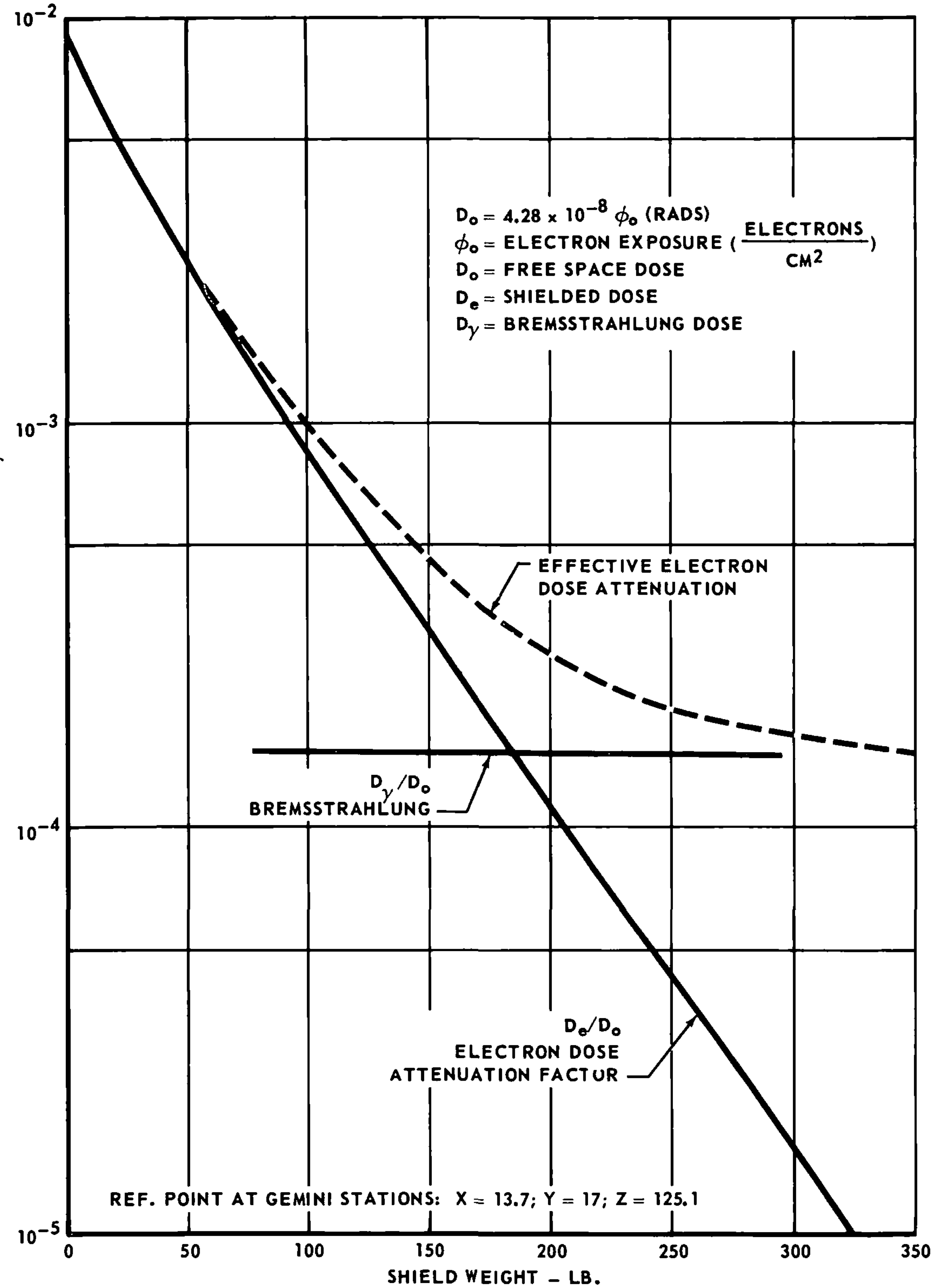


FIGURE 8.8-7

DOSE PER UNIT FLUX OF PROTONS OF ENERGY > 30 MEV

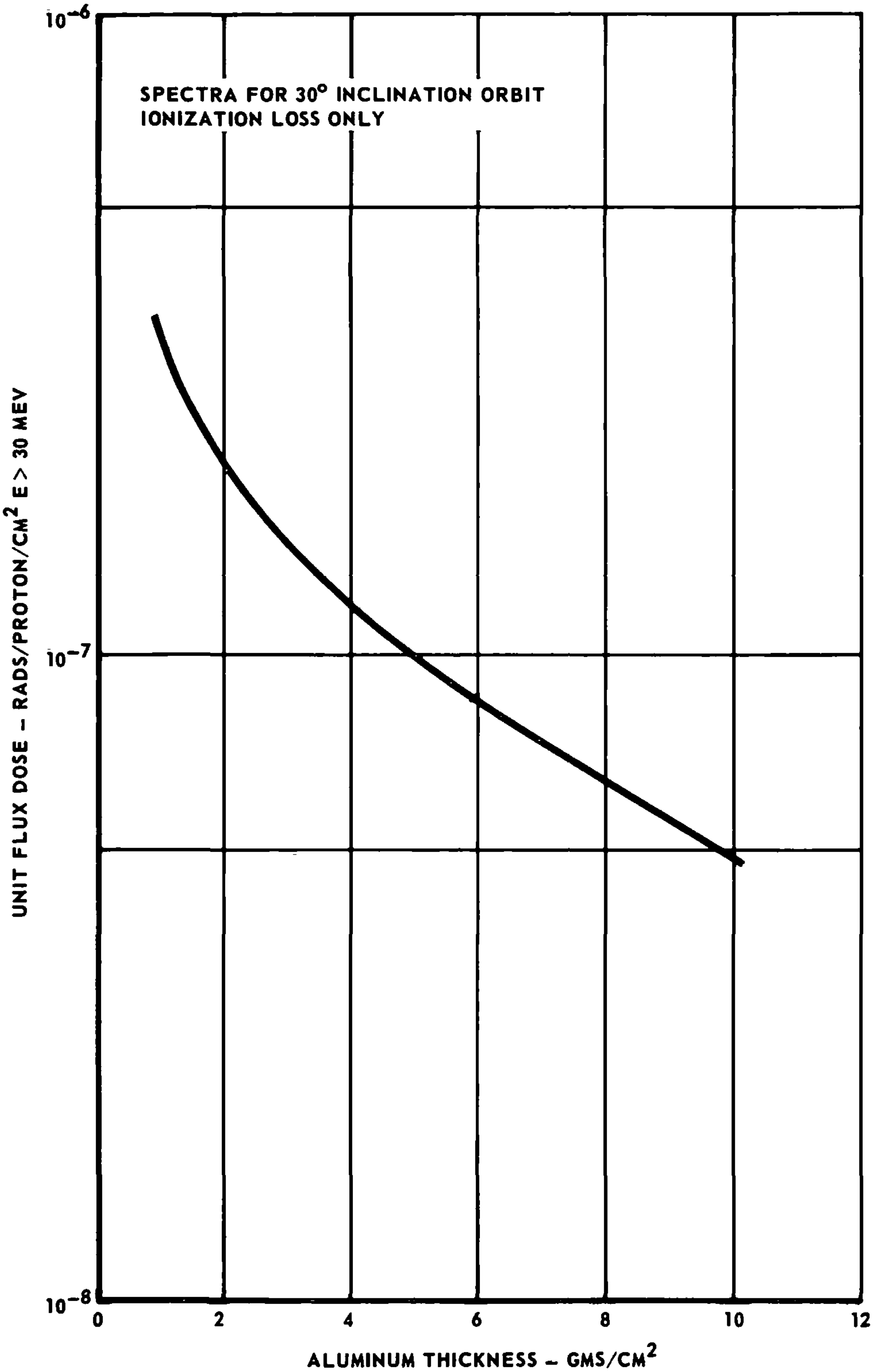


FIGURE 8.8-8

8.8.2 (Continued)

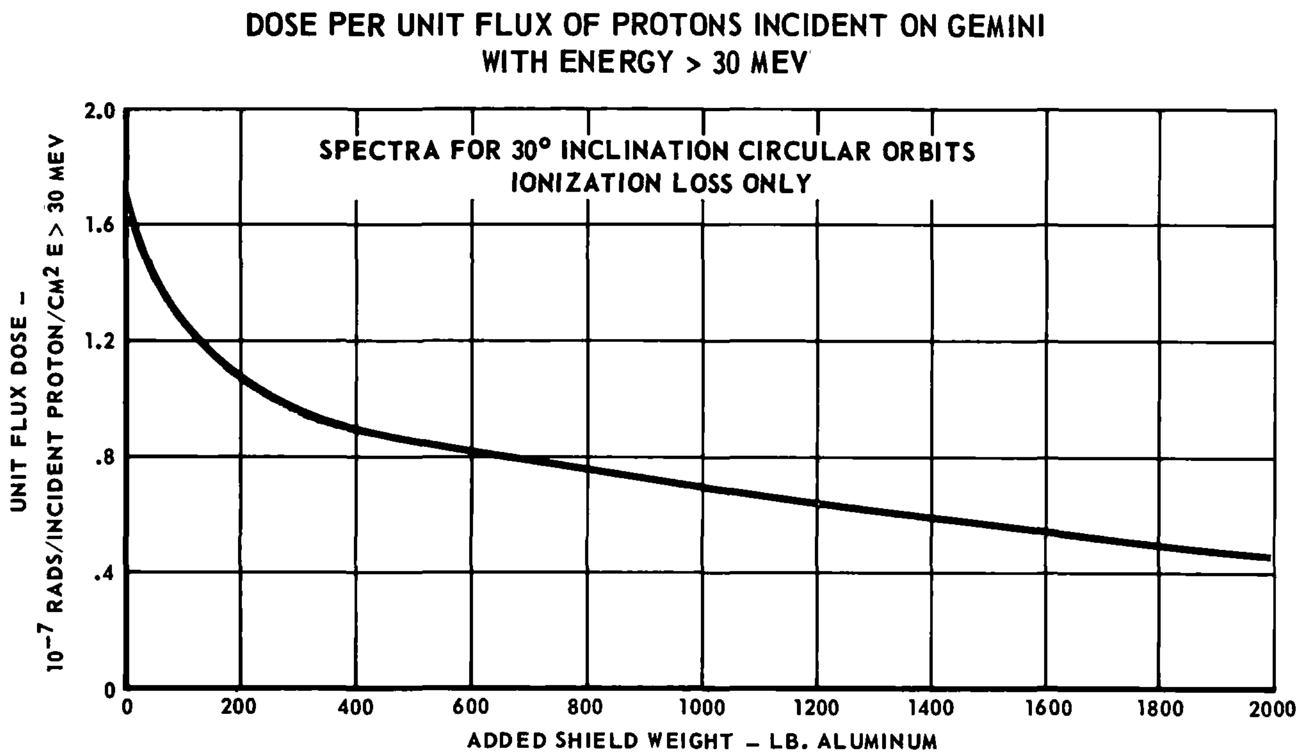


FIGURE 8.8-9

TABLE 8.8-1

SURFACE DOSE PER UNIT INCIDENT FLUX
FERRY INHERENT SHIELDING

ELECTRONS	BREMSSTRAHLUNG	PROTONS
(RAD/ELECTRON/CM^2) 3.9×10^{-10}	(RAD/ELECTRON/CM^2) 6.7×10^{-12}	(RAD/PROTON/CM^2) 1.6×10^{-7}

8.8.3 Allowable Dose - The allowable doses used as criteria for evaluating crew rotation operations are contained in Reference 8.8-6. The suggested average yearly dose is 233 rad for skin or surface dose to the whole body, 54 rad to the blood forming organs at an average depth of 5 cm and 27 rad to the eyes at a depth of 3 mm.

8.8.3 (Continued)

The optimization of shield design for the ferry mission is an interface problem depending on the set of total allowable doses finally selected and on the operational effects associated with the addition of shielding to the Ferry and space station (e.g., spacecraft weight compared to launch vehicle capability). An analysis of this nature is properly the subject of subsequent effort. The results would include an apportionment of the allowable dose values between the Ferry mission and the space station mission.

As an example case, it is assumed that 10% of the total allowable dose is allowable for the Ferry mission phase. A factor of safety of four is applied to compensate for uncertainties in calculated shielding effectiveness and execution of shield design.

Design allowable doses for the Ferry mission resulting from these assumptions are 6 rads for surface dose, 1.4 rads for the blood forming organs and 0.7 rads for the eyes. For emergency operations allowable doses were taken as 100% of the maximum single acute exposure suggested in Reference 8.8-6. With a factor of safety of 4, allowable emergency doses are: 125 rad for surface, 50 rad for the blood forming organs and 25 rad for the eyes.

8.8.4 <u>Doses with Inherent Shielding Only</u> - A typical mission consists of the phases and times shown in Figure 2.3-1. The period the crew is outside the Ferry or space station has a major effect on total dose. When in open space, inherent shielding is that provided by the space suit and extravehicular coverall. Normally, crew open space transfers will be made quickly during periods outside of the regions of intense radiation. Emergency operations may necessitate transfer while in the high-intensity region.

Doses received, with inherent shielding, in a mission to a station in a 250 na. mi. altitude orbit are shown in Table 8.8-2. Surface doses are assumed

8.8.4 (Continued)

to apply uniformly over the body surface. For blood forming organs the additional 5 gm/cm^2 shielding results in a negligible electron dose but has a relatively small effect on the bremsstrahlung and proton doses. For simplicity, the sum of the surface proton and gamma ray doses is taken as the dose received at 5 cm depth.

TABLE 8.8-2

RADIATION DOSE SUMMARIES

250 NA.MI. – 30° INCLINATION CIRCULAR ORBIT
INHERENT SHIELDING PLUS 1 $LB./FT.^2$ SUIT SHIELDING

MISSION PHASE	TIME (HOURS)	TYPE	SURFACE DOSE (RAD.)	
			16 JULY 1962 ENVIRONMENT	PROJECTED JULY 1967 ENVIRONMENT
CATCH-UP – RENDEZVOUS – DOCK AND MOOR	23.1	D_e	24.6	0.13
		D_γ	1.62	0.008
		D_P	0.15	0.15
EXTRAVEHICULAR TRANSFER (2 TRIPS – 1/2 HR. EACH)	1.0	D_e	9.2–1400	0.72 – 13.2
		D_γ	0	0
		D_P	0.01 – 0.35	0.01 – 0.35
SEPARATE PARK AND RE-ENTRY	4.0	D_e	0.34 – 22.5	0.027 – 0.13
		D_γ	0.023 – 1.48	0.002 – .008
		D_P	0.014 – 0.11	0.014 – 0.11
TOTALS	28.1	D_e	34.1 – 1447	0.88 – 13.48
		D_γ	1.6 – 3.1	0.01 – 0.02
		D_P	0.2 – 1	0.17 – 0.61
TOTAL DOSE			35.9 – 1451	1.06 – 14.11

If the predicted 1967 environment prevails, normal and emergency ferry operations can be performed to stations at altitudes up to 250 na. mi. with received doses within the allowable without added shielding other than a small amount of local shielding for the eyes.

8.8.4 (Continued)

If the 16 July 1962 environment prevails, normal and emergency ferry operations cannot be performed with received doses within the allowable rate without either a substantial amount of added radiation shielding or reduction in space station orbit altitude below 250 na. mi. To meet the dose limits during a ferry mission, additional shielding would be required to provide the following additional dose attenuation factors:

Normal ferry operations in low intensity regions

Skin, whole body, 1/6; Blood forming, 1/1.3; and Eyes, 1/51.4

Emergency operations in high intensity regions

Skin, whole body, 1/11.6; Blood forming, 1/1; and Eyes, 1/58

It is indicated in Figure 8.8-1 that with the eyes adequately protected, radiation doses will be reduced to within the allowable by reducing the station orbit altitude to approximately 160 na. mi. If the eyes were not protected, reduction of station orbit attitude to 105 na. mi. would be necessary.

8.8.5 Added Shielding for Ferry Missions - The results of calculations of doses with various amounts of added shielding for normal operations (minimum exposure) and emergency operations (maximum exposure) are presented in Figure 8.8-10 and 8.8-11, respectively.

As an example, the breakdown of the ferry mission dose into components is shown in Table 8.8-3 for 200 lbs. shield weight added to the spacecraft, and 70 lbs. per crew member added personal shielding during transfer. Comparison with the inherent shielding case emphasizes the shift in the relative contributions of electrons, bremsstrahlung and protons as shielding is added.

Eye Shielding for 1967 Predicted Environment - 250 Na. Mi. Station Orbit - A 30 percent attenuation of radiation necessary to reduce eye dose to the allowable for both normal and emergency operations is easily obtained with space helmet (plastic) face plates of 0.29 inches total thickness (1.5 psf).

EFFECT OF ADDED SHIELD WEIGHT ON TOTAL MISSION DOSE

MINIMUM EXPOSURE CONDITIONS

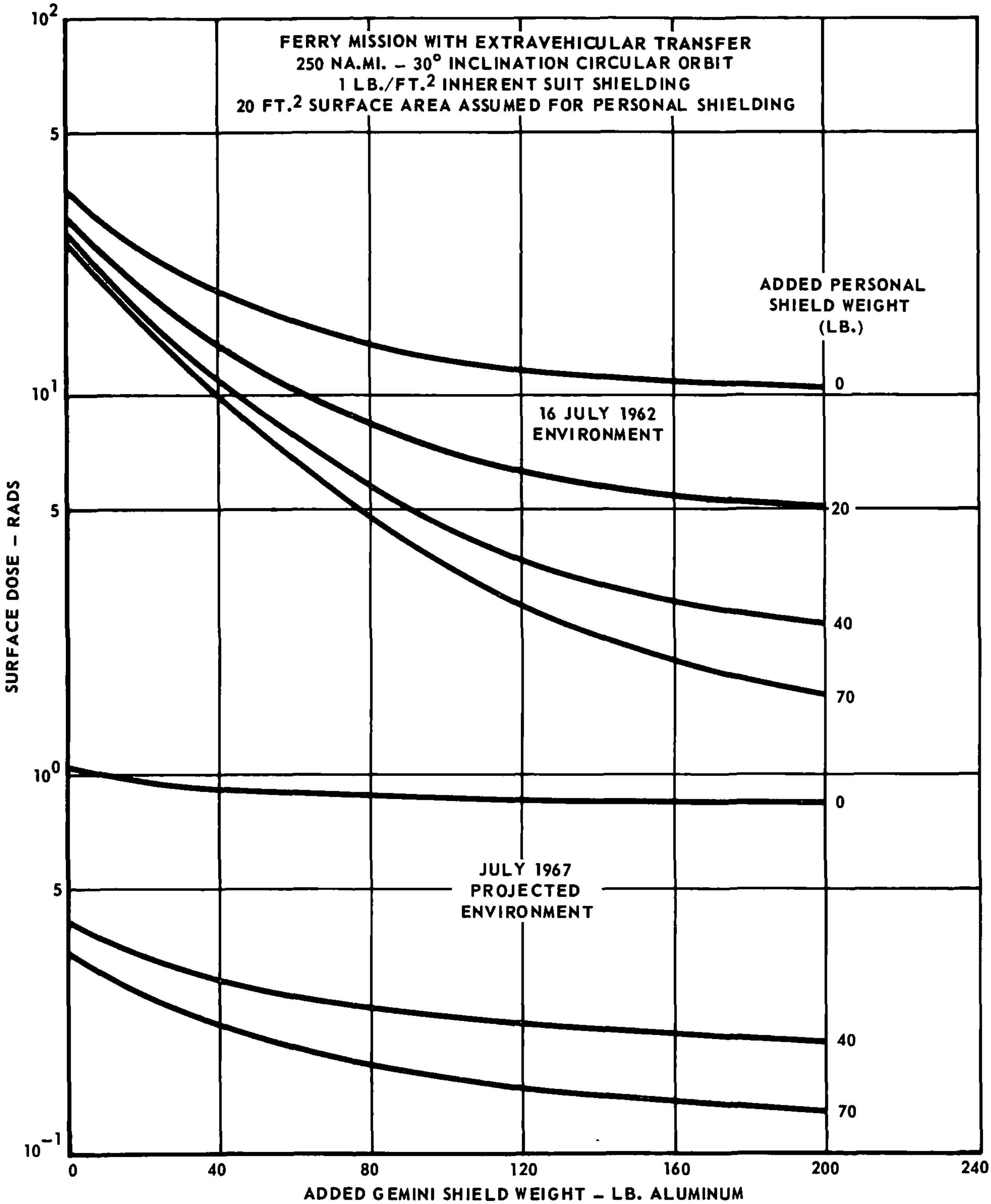


FIGURE 8.8-10

EFFECT OF ADDED PERSONAL SHIELD WEIGHT ON TOTAL MISSION DOSE

MAXIMUM EXPOSURE CONDITIONS – FERRY MISSION WITH EXTRAVEHICULAR TRANSFER

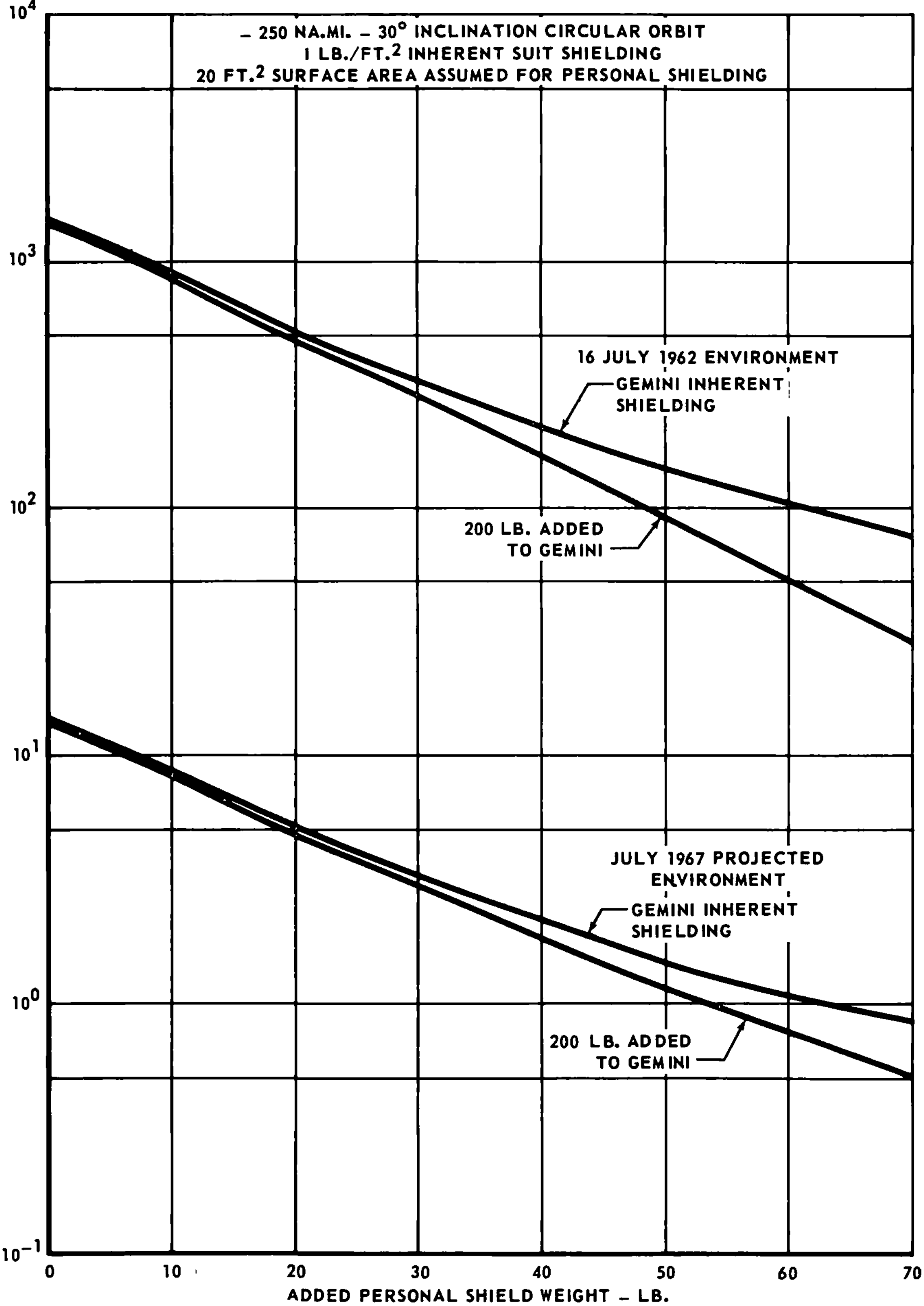


FIGURE 8.8-11

8.8.5 (Continued)

TABLE 8.8-3
RADIATION DOSE SUMMARIES

250 NA.MI. – 30° INCLINATION CIRCULAR ORBIT
200 LB. SHIELD WEIGHT ADDED TO GEMINI
PLUS 70 LB. PERSONAL SHIELD WEIGHT
ADDED TO EACH CREW MEMBER DURING TRANSFER

		SURFACE DOSE (RAD.)	
	TYPE OF RADIATION	16 JULY 1962 ENVIRONMENT	PROJECTED JULY 1967 ENVIRONMENT
TOTAL EXCLUDING TRANSFER	ELECTRON GAMMA PROTON	0.295 – 0.556 1.64 – 3.10 0.108 – 0.171	0.0019 – 0.0031 0.010 – 0.017 0.108 – 0.171
EXTRAVEHICULAR TRANSFER	ELECTRON GAMMA PROTON	0.171–26.3 0.00 – 0.0 0.00 – 0.086	0.0133 – 0.245 0 – 0 0 – 0.086
TOTALS	ELECTRON GAMMA PROTON	0.466 – 26.86 1.64 – 3.10 0.108 – 0.26	0.0152 – 0.248 0.010 – 0.017 0.108 – 0.257
TOTAL DOSE	ALL	2.214 –30.22	0.133 – 0.522

Body Shielding Required for 16 July 1962 Environment - 250 Na. Mi. Orbit - To provide adequate protection for emergency operations, primarily an open-space exposure, minimum added shielding weight is obtained by providing personal body protection sufficient for emergency operations together with spacecraft shielding sufficient for normal operations.

With this technique, approximately 2.75 lbs. per square foot added personal body shielding is needed for one hour exposure. This could conveniently be incorporated into the material of a space suit coverall. A coverall of about 0.5 inch thick rubberoid material would suffice and, in addition, would provide meteoroid protection. Such coveralls would weigh about 55 pounds per man, based on a 1 hour transfer time. If transfer time were reduced to 15 minutes, 20 pounds per man would be needed.

With the appropriate amount of personal protection, about 75 pounds of aluminum spacecraft shielding, as shown in Figure 8.8-12, reduces total body

ISODOSE CONTOURS FOR THE GEMINI FERRY MISSION WITH EXTRAVEHICULAR TRANSFER

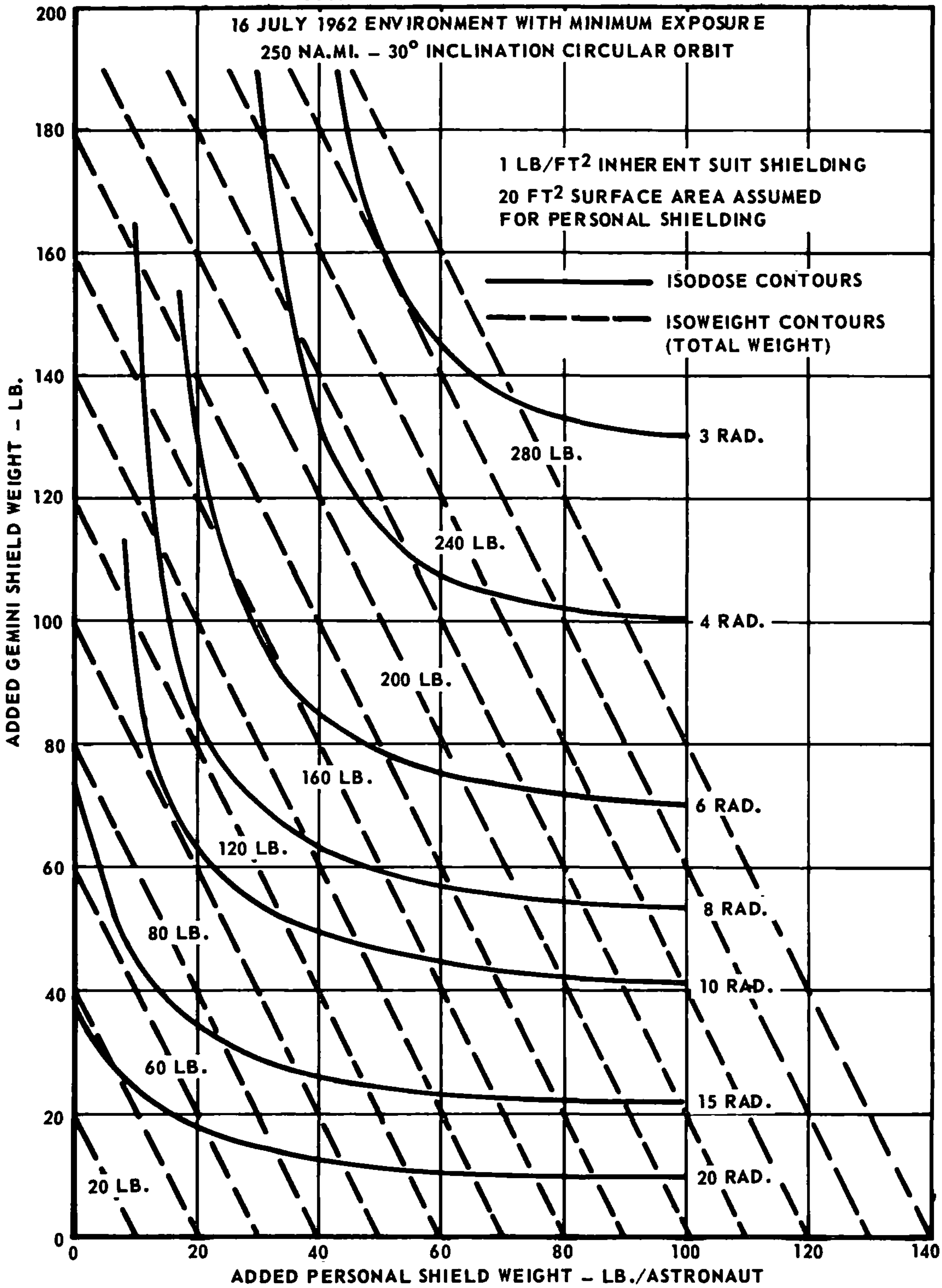


FIGURE 8.8-12

8.8.5 (Continued)

dose to the allowable in normal operations. The total added weight is 185 lbs.

If only normal operations were considered, the minimum additional weight required is 30 lbs. per man personal shielding plus 95 lbs. spacecraft shielding, or a total of 155 lbs. Thus allowing for emergency operations produces a weight increase of only 30 lbs.

Eye Shielding for 16 July 1962 Environment - 250 Na. Mi. Orbit - With 75 lbs. added to the Ferry spacecraft the emergency conditions may be satisfied by covering the eyes with a local shield of about 2.5 gm/cm^2 (0.8 inches thick) plexiglass. The dose received under normal operating conditions would be approximately 1.64 rads resulting almost entirely from gamma radiation. However, consideration of self shielding (which was not included in bremsstrahlung and proton calculations) would reduce the dose to the allowable. If self shielding is neglected about 0.26 inches of leaded glass (4 gms/cm^2) would be needed.

Discussion - This investigation indicates that radiation dose levels encountered during the ferry mission can be maintained within allowables without modification to the spacecraft and without serious operating restrictions for the 1967 environment.

In the event high attitude nuclear testing is resumed, and an environment similar to the 16 July 1962 environment is produced, ferry missions to a 250 na. mi. station altitude are feasible with the addition of about 200 pounds total of personal and spacecraft shielding.

9. STANDBY IN SPACE

Based on MORL scheduling, some of the Ferry spacecraft remain moored to the station in a deactivated condition for six to eight months. Analysis of the existing Gemini capability to meet this criteria has proceeded according to the following general plan:

A. Determination of the extremes of the space environment and the associated hazards.

B. Analysis of the normal standby capability, reliability and confidence level for existing Gemini equipment.

C. Determination of the modifications to the design and operating procedures (including partial and complete encapsulation of the spacecraft) needed to provide protection during standby.

D. Determination of the minimum operable systems necessary to insure safe return to earth in order to provide special protection for these systems.

9.1 Hazards - The hazards of long standby result primarily from the temperature, radiation, vacuum, ultraviolet and meteoroid environments at the space station orbital altitude and inclination.

9.1.1 Temperature Environment - Extreme low or high temperatures present a standby hazard to equipment, and possibly to structure. Two factors contribute to the extreme temperatures experienced by the Ferry Spacecraft during standby: (1) the spacecraft is sun-oriented, since the MORL uses solar cells for electric power and, (2) powered equipment does not operate during standby; hence, no internal heat is generated.

Factors determining spacecraft temperatures include the angle between the sun's rays and the spacecraft orbit, earth emitted heat, spacecraft orientation, the thermal radiation characteristics of spacecraft external and internal surfaces, structural heat conduction and internally generated heat.

9.1.1 (Continued)

Gemini uses high infrared emissivity coating on external surfaces of the re-entry module in order to radiate large amounts of heat during launch and re-entry. In order to obtain a high radiator efficiency during the orbit phase, a low solar absorptivity coating is used on the adapter. During periods of low internal heat generation, however, the adapter equipment becomes too cold with this radiator coating. To alleviate this cooling, the adapter aft cover is provided with a coating having a high ratio of solar absorptivity to infrared emissivity, thus adding heat when the adapter cover is oriented towards the sun.

When the Ferry Spacecraft is oriented so that no direct sunlight falls on the adapter aft cover, however, temperatures become too cold. For analysis the orientation yielding the lowest adapter temperature was chosen. Conditions assumed were: (1) the aft cover is oriented to deep space at the noon orbital position of the spacecraft (longitudinal axis normal to sun's rays), (2) the station is not spinning, and (3) the orbit plane is 60° from the sun's rays. A non-spinning spacecraft is considered for the analysis, since temperatures are lower for this condition. Temperatures at other spacecraft mooring positions and orientations are referenced to this analysis.

External heat inputs to the spacecraft for the assumed orientation were determined for four areas around the circumference at several longitudinal locations. These were used to calculate temperatures of major internal components. Temperatures are summarized in Table 9.1-1.

The adapter temperature of a spacecraft which is moored with the aft cover 180° from the sun's rays is slightly warmer than shown in the table, since some reflected heat from earth will be incident on the aft cover. The re-entry module temperatures of these spacecraft, however, may be lower, due to shadowing of the surfaces by the MORL.

9.1.1 (Continued)

TABLE 9.1-1
TYPICAL FERRY SPACECRAFT TEMPERATURES DURING STANDBY

COMPONENT	TEMPERATURE RANGE - °F
AFT COVER	-60 TO -55
RADIATOR SURFACE	-103 TO +1
EQUIPMENT IN ADAPTER	-56 TO -54
RE-ENTRY MODULE SURFACE	-156 TO +255
EQUIPMENT IN RE-ENTRY MODULE	-65 TO -16

High adapter temperatures occur in a spacecraft which is nose moored with the aft cover towards the sun. For this orientation, and the present aft cover coating, the maximum equipment temperature is 395°F, with the aft cover rising to 427°F.

The coatings on the Unmanned Supply Spacecraft and on the cargo module of the Ferry/Supply Spacecraft are chosen to provide passive temperature control during the standby period, as well as during the catch-up orbits. The choice of these coatings is discussed in Section 9.3.1.

9.1.2 Radiation Environment - The radiation hazard to the equipment on board the Gemini Ferry is principally due to the artificial electrons produced in the 9 July 1962 high altitude nuclear explosion (Starfish) and those which may be produced in subsequent shots.

Assuming an additional explosion, identical in effect to the 9 July 1962 shot, occurring immediately before launch, and subsequently decaying in the predicted manner the maximum total unshielded exposure for the standby 6-months period at 250 na. mi. altitude is $\phi = 2.4 \times 10^{13}$ electrons/cm^2, corresponding to an absorbed dose of 9.8×10^5 rads. Based on the best estimate of electron flux, spectrum (fission Beta spectrum), and decay with time, (assuming no further explosions) the projected 1967 unshielded standby dose is 2×10^4 rads.

9.1.3 Meteoroid Environment - The many estimates of the meteoroid flux and theories of penetration mechanism produce a broad band of predicted results. Inherent meteoroid puncture protection afforded by the basic Gemini structure has been evaluated for three distribution-penetration theory combinations. Penetration frequency for aluminum targets, based on Whipple 1963, Whipple 1961, and NASA-Gemini distributions and Summers and modified Bjork penetration theories are shown in Figure 9.1-1. Since more recent experimental and theoretical data are incorporated in Whipple's 1963 distribution, this is believed to be the best estimate of meteoroid environment. Recent data from Explorer XVI also partially confirms this belief.

With the '63 distribution, both the Summers and modified Bjork penetration equations are shown to yield approximately the same result for an aluminum target. This peculiarity is due to Whipple's assumed values for meteoroid density and velocity, ρ_m = .44 gm/cc and V = 22 km/sec, respectively. Since experimental data on penetration of structural materials by particles traveling at meteoritic velocities does not exist, neither penetration equation has been verified. Bjork's equation has been used to analyze structural resistance to penetration since limited tests on impacts in lead at Mach numbers greater than one tend to verify his theoretical analysis that the exponent of velocity ratio V_m/C_T should be one-third. This analysis, however, was for a projectile and target of the same material, and therefore the equation is modified to account for different materials by including a density ratio $(\rho_m/\rho_t)^{1/3}$. The one-third exponent is a pessimistic estimate as long as ρ_m is less than ρ_t.

The meteoroid hazard to the Ferry Spacecraft for the three different environment-penetration estimates and 230 days exposure is presented in Figure 9.1-2. Penetrations into equipment bays and the pressure cabin are listed separately.

ALUMINUM METEOROID PENETRATION FREQUENCY

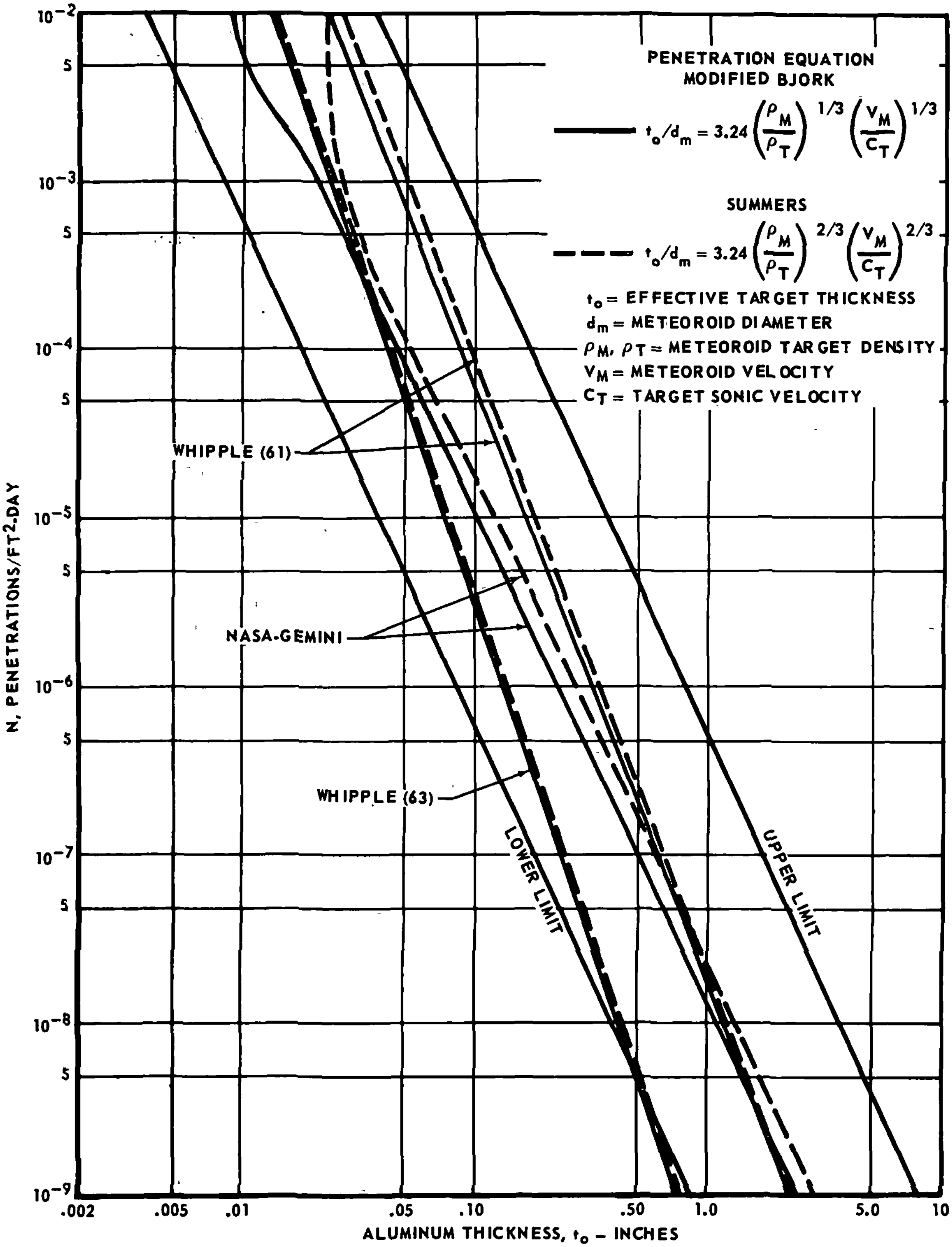
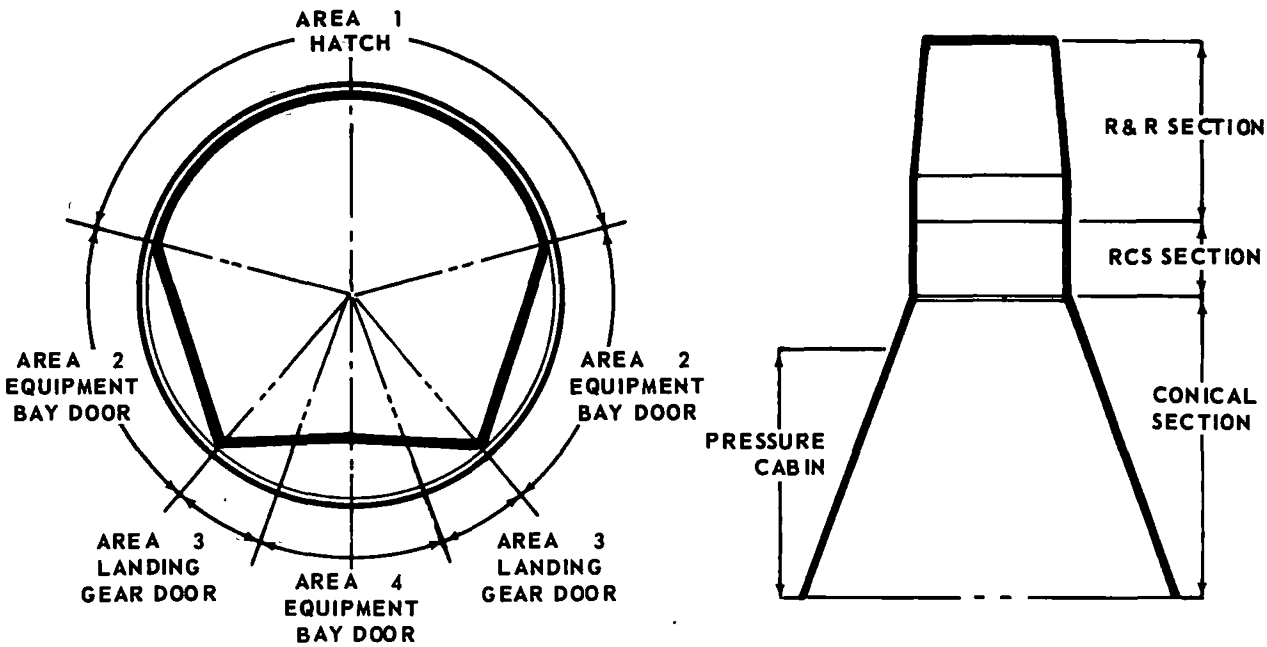


FIGURE 9.1-1

METEOROID HAZARD EVALUATION
GEMINI RE-ENTRY MODULE

PRESSURE CABIN CONFIGURATION



	LOCATION \ THEORY	PROBABILITY OF NO PENETRATION: WHIPPLE '63 & BJORK	WHIPPLE '61 & BJORK	NASA GEMINI & SUMMER
	RE-ENTRY MODULE	.9993	.9800	.9941
CREW PROTECTION	PRESSURE CABIN	.9998	.9930	.9981
	AREA 1	.99985	.9952	.99882
	AREA 2	.99994	.9985	.99960
	AREA 3 – FWD.	.99999	.99992	.99997
	AREA 3 – AFT	.99999	.99990	.99997
	AREA 4 – FWD.	.99999	.99997	.99999
	AREA 4 – AFT	.99998	.99958	.99990
	CABIN WINDOWS	.99999	.99973	.99989
EQUIPMENT PROTECTION	EQUIPMENT BAYS	.9996	.9870	.9960
	R & R SECTION	.99990	.99917	.99980
	RCS SECTION	.99999	.99973	.99922
	CONICAL SECTION:			
	FWD. EQUIP. BAY	.99990	.99720	.99937
	EQUIP. BAY DOORS	.99980	.99420	.99869
	L.G. DOORS – FWD.	.99995	.99870	.99972
	L.G. DOORS – AFT	.99996	.99880	.99972
	ECS DOOR	.99998	.99947	.99988

FIGURE 9.1–2

9.1.3 (Continued)

For all environments, the probability of no penetration into the pressure cabin is slightly higher than for equipment bays. Whipple's '63 distribution yields .9993 probability of no penetration, the highest value for the three environments considered.

The meteoroid hazard to a Gemini Spacecraft assuming all systems on board are required to operate is shown in Figure 9.1-3 as a function of time in orbit. This assumption corresponds to the hazard to Gemini that would exist if there were sufficient expendables on board to orbit for the indicated times.

The hazard to the Ferry Spacecraft considering all systems to be needed through docking and considering only those used after standby is also shown. The hazard to the Ferry Spacecraft for 230 days is lower than to Gemini for 14 days. This exemplifies the powerful effect on probability of success of reducing the number of systems required to operate post standby, e.g., the fuel cells, radiator, orbit attitude and maneuvering system and primary oxygen supplies are not used post standby.

9.2 <u>Effects of Standby on Spacecraft Systems</u> - Available studies and test data on the effects of prolonged exposure to the space environment are limited. However, reliable operation of ferry equipment and systems after long inoperative standby appears attainable. Equipment which cannot be redesigned for prolonged storage or passively protected against deterioration in space can be actively protected by heaters, continuous pressurization and, in some cases, by periodic operation to reverse a deteriorating trend.

Cold welding of metals might occur in unpressurized areas where metals are in contact, such as in mechanical systems and valves, depending on the ambient pressure, the temperature, the type of metals in contact, the nature of the contact surfaces and the bearing pressure. Cold welding can be prevented, or undesirable

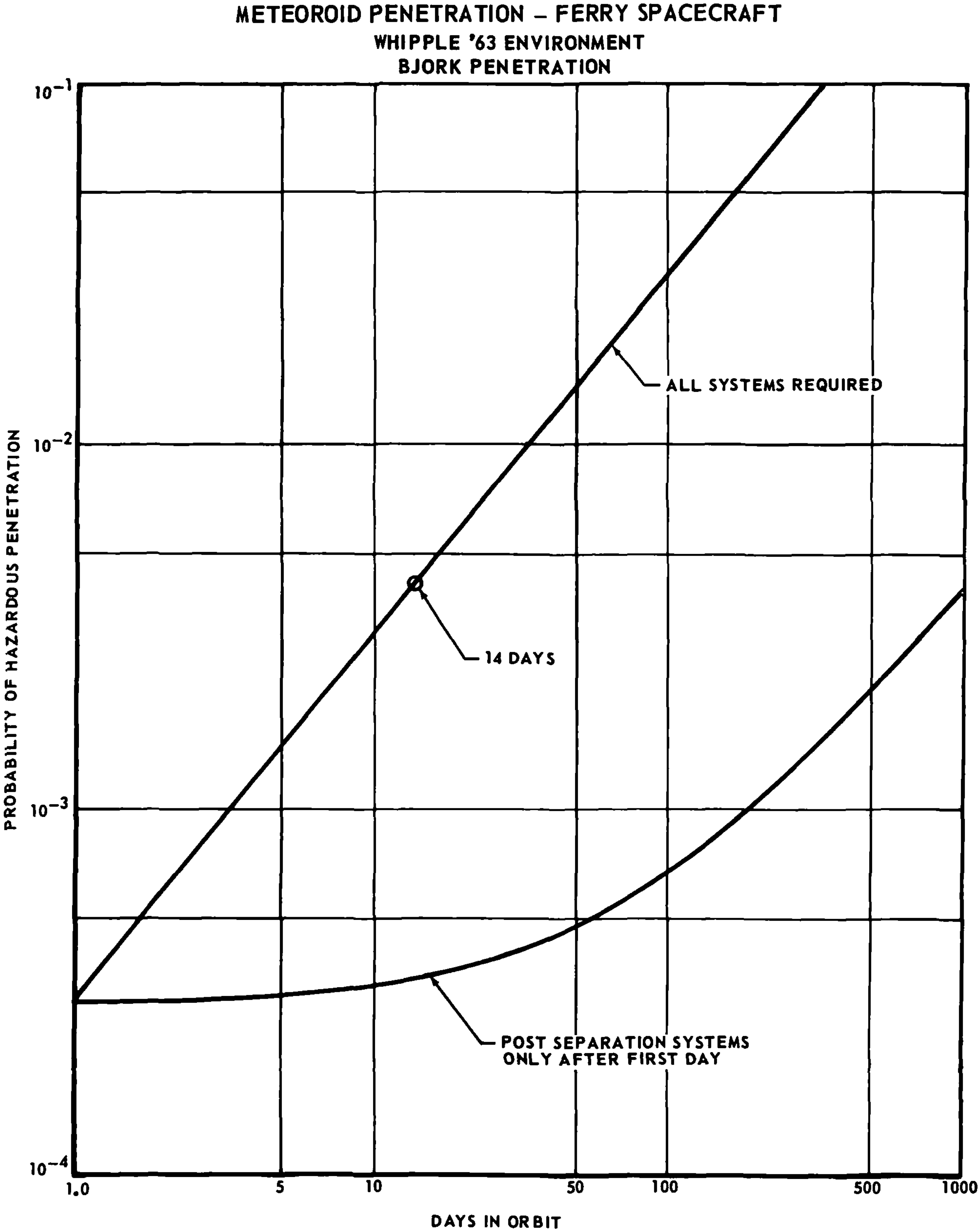

METEOROID PENETRATION – FERRY SPACECRAFT
WHIPPLE '63 ENVIRONMENT
BJORK PENETRATION
PROBABILITY OF HAZARDOUS PENETRATION
10^{-1}
10^{-2}
10^{-3}
10^{-4}
ALL SYSTEMS REQUIRED
14 DAYS
POST SEPARATION SYSTEMS
ONLY AFTER FIRST DAY
1.0
5
10
50
100
500
1000
DAYS IN ORBIT

FIGURE 9.1-3

9.2 (Continued)

effects reduced, by: (1) pressurization, (2) proper lubricants, and (3) using hard and insoluble dissimilar contact surfaces.

The most critical effect of high-energy particle radiation in space is the degradation in performance of elastomeric compounds. The sensitivity of different elastomers varies widely. Butyl rubber compounds lose nearly all utility when exposed to a total dose in the 10^5 to 5×10^6 rad range. In the 10^6 to 10^7 rad range, other rubbers progressively lose from 5 to 25 percent of their original tensile strength and ultimate elongation. In addition, thermal cycling and stressing of the elastomers accelerates the adverse effects.

The principal radiation damage mechanisms are cross-linking, chain scission, and molecular rearrangement in the elastomer material. Stress-strain properties (tensile strength, ultimate elongation) and mechanical properties (dynamic modulus, hysteresis, compression and permanent set, deformation load) are especially subject to deterioration, primarily by cross-linking. The radiation effect on elastomer performance is minimized by selection of materials which possess relatively high radiation damage thresholds.

Environmental effects unique to each system are discussed in the following sections.

9.2.1 Electrical Equipment - Both the power source and power distribution components of the electrical power system used during crew return and recovery are susceptible to the hazards of extended space environment including temperature, radiation and vacuum.

Temperature - The remotely automatically activated batteries used for post-separation power may attain a stabilized temperature of approximately -50°F during the standby period if no active temperature control is provided. No physical

9.2.1 (Continued)

deterioration of the battery is expected at this temperature, although it approaches the -65° limit specified in military and spacecraft design specifications presently in use. Similarly, the stored electrolyte does not freeze above -80°F for the KOH concentration selected.

In order to obtain rated output, however, the battery operating temperature must approximate +80°F (Figure 9.2-1). The operating temperature can be attained by using integral electric heaters, powered from the MORL power system, either continuously or just prior to activation.

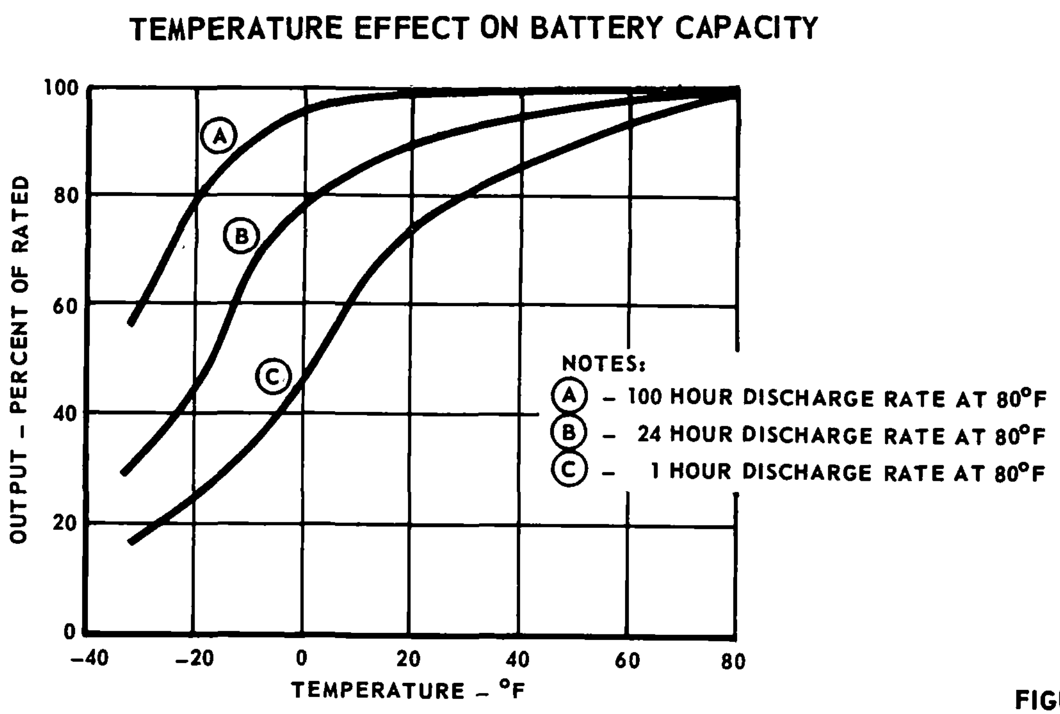


FIGURE 9.2-1

Continuous active temperature control uses valuable power from the space station supply, however, and if heat-up is delayed until activation, the time required to attain operating temperature would delay emergency departure procedures. Another approach is the use of pyrotechnic heaters around the electrolyte manifold. The activation signal simultaneously ignites the gas generator and pyrotechnic heater, which heats the electrolyte before it is injected into the battery cells. This method requires little MORL electrical energy and provides rapid heating.

9.2.1 (Continued)

Power distribution system components are not expected to be affected by the orbital temperature conditions although any components exposed to temperatures below -65°F would need additional testing.

Radiation - Exposure to ambient radiation is not expected to affect either the power sources or the distribution system. Investigations of radiation effects on batteries show that the cell cases and separators are the components most susceptible to damage. For the anticipated total integrated dose, standard acrylic cell cases and cellulose separators appear adequate; however, the radiation damage threshold of these components can be increased by at least two orders of magnitude through the use of different materials. These materials such as polystyrene cell cases and non-woven separators are used in "radiation hard" batteries presently operating. No information appears to be available on radiation effects on pyrotechnic gas generators for the automatically activated batteries.

Ground tests of radiation effects on power distribution system components, such as wire, connectors, potting compounds, semi-conductors, and relays, show that proper selection of materials will avoid damage for the expected doses.

Vacuum - Samples of Gemini wire are currently undergoing tests to determine the effects of exposure to simulated space vacuum on current carrying capacity and on physical characteristics during the longest Gemini mission. Since the ferry mission is considerably longer, additional endurance tests are needed.

Gemini-type toggle and rotary switches and circuit breakers need additional testing since these components are not hermetically sealed. Contact contamination, due to out-gassing of materials in the spacecraft and possible cold welding, should be investigated.

Critical Battery Shelf Life - The manually activated batteries, used during the pre-standby phase of the ferry mission, all reach the critical activated life

9.2.1 (Continued)

point during standby. At this point, failure of the cell separators due to exposure to electrolyte and dissolved metals leads to internal shorts accompanied by the evolution of gases and heat. The Gemini battery vendor estimates that such evolution does not occur if less than 10 percent of capacity remains in each battery at the critical life point. However, the use of redundant batteries results in more than 10 percent capacity remaining, if there are no battery failures. The effects are alleviated by the low battery temperature during standby (-50°F) which reduces the chemical activity. Nevertheless, positive means to completely discharge the remaining capacity at a low rate after arrival at the MORL, may be needed to eliminate gassing or overheating.

9.2.2 Electronic Equipment

Temperature Effects - In general, Gemini electronic equipment is specified to be operable in an ambient temperature range of -60°F to +160°F. No damage will occur during standby provided these extremes are not exceeded, with one exception; the inertial platform gyros and accelerometers mounted on the platform stable element are protected from damage caused by accidental contact of the inertial elements with their stops during standby by a gel which liquifies above 100°F. Thus, the platform storage temperature should be maintained below 100°F. Prior to operation the gel is warmed above 100°F and acts as the flotation fluid.

Radiation Effects - Energetic charged particles, the principal component of space radiation, produce ionization and crystal lattice disturbances in electronic components. These disturbances induce changes such as increased conductivity in semiconductors, polymerization to the point of gellation of hydrocarbon lubricants and gyro fluids, and breakdown of dielectrics and seals. As a general rule the greater the conductivity of a semiconductor device, the greater its radiation resistance. Thus silicon components are more susceptible than germanium components,

9.2.2 (Continued)

and thick base, power switching transistors are more easily damaged than thin base, high frequency transistors. Tunnel diodes are affected by only the highest radiation levels.

When gas within a transistor enclosure is ionized by radiation, performance is adversely affected by a temporary surface ionization. This degradation is greatly accelerated if the transistor is reverse-biased, the effect increasing with the square of the bias voltage. Transistors used in digital circuitry such as in the Gemini Computer and Digital Command System, are particularly sensitive. The total radiation dose, rather than the dose rate, determines the amount of degradation that occurs. Self-healing takes place when power is removed, although, once damaged, transistors appear more susceptible to further radiation damage. This surface effect can be avoided by using suitable transistor manufacturing techniques and by aging the transistors in a radiation field to determine sensitivity.

Nuclear explosions also produce a radiation hazard. At ranges up to a few hundred miles from an extra-atmospheric explosion there will be a short pulse of neutrons and gamma radiation of extremely high dose rate. The pulse duration, however, is less than a millisecond; therefore, the integrated dose damage threshold will not be exceeded. Permanent damage from high dose rates occurs only at ranges at which thermal damage would predominate. Therefore, the integrated radiation dose is assumed to uniquely determine electronic component sensitivity.

Radiation susceptibility of various electrical and electronic equipment is shown in Figure 9.2-2. The calculated value, 9.8×10^5 rads received dose, during standby is lower than the minimum threshold for all but one item, elastomers. For most of the sensitive components protection will be provided by the Ferry Spacecraft skin (approximately 1.4 lb/ft^2), equipment casing, and other components. Considering only the shielding afforded by the spacecraft

9.2.2 (Continued)

RADIATION SUSCEPTIBILITY OF ELECTRICAL
AND ELECTRONIC EQUIPMENT

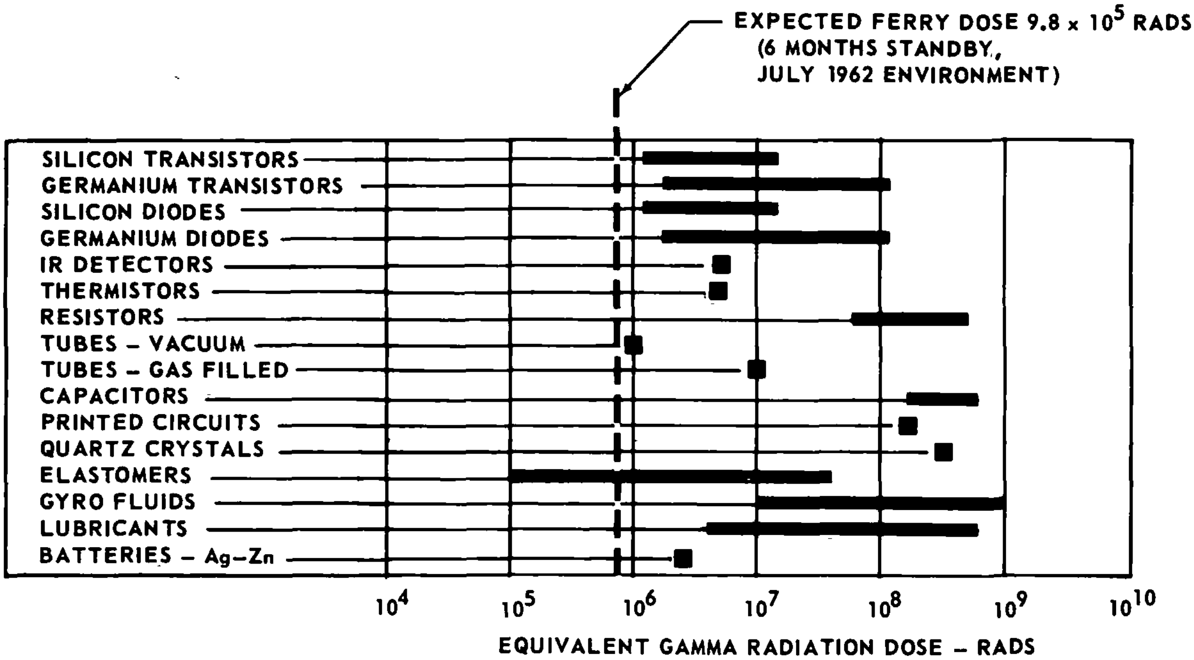


FIGURE 9.2-2

skin, the value of 9.8×10^5 rads is reduced to approximately 2×10^5 rads. The remainder of the spacecraft will provide an additional attenuation factor which will approach 0.5 (2π shielding) for many of the items. Further reduction would require only small amounts of additional shielding.

Vacuum Effects - Prolonged exposure to low ambient pressure causes evaporation of lubricants and outgassing of seals, insulators, and potting compounds. For a six month exposure, however, these effects are expected to be minor for the materials used in Gemini. The most serious problem is loss of pressurization in pressurized components. Components and systems having high electric fields, such as coaxial switches, multiplexers and the radar tracking beacons, require pressurization to prevent arcing. For standby capability these components should be hermetically sealed or repressurized before activation.

9.2.2 (Continued)

Inertial platform pressurization prevents contamination and avoids material problems that would occur in a hard vacuum. When platform internal pressure falls below 0.3 psia, the gas thermal conduction properties begin to degrade. These properties are important during re-entry when the platform must operate with only heat sink cooling. Since platform internal pressure will fall to 0.3 psia in approximately four and one-third months, repressurization appears necessary.

The sensitivity of Gemini electronics to vacuum effects is summarized in Figure 9.2-3.

9.2.3 Propulsion Equipment - Materials used in the Gemini Orbit Attitude and Maneuver System and Re-entry Control System are compatible with the propellants and no degradation in performance or ignition characteristics is anticipated after six to eight months of space storage.

The retrograde rocket motors are being developed to operate satisfactorily after 14 days in a hard vacuum at temperatures from -20° to +180°F. The manufacturer

VACUUM EFFECTS ON GEMINI ELECTRONIC EQUIPMENT

SYSTEM/COMPONENT	EFFECT	FIX
GUIDANCE AND CONTROL		
HORIZON SENSOR HEADS	LUBRICANT LOSS	REDESIGN BEARINGS (IF NECESSARY)
INERTIAL PLATFORM	PRESSURE LOSS IMPAIRS HEAT TRANSFER	REPRESSURIZATION
RENDEZVOUS RADAR	(NOT REQUIRED AFTER RENDEZVOUS)	
COMMUNICATIONS		
HF & UHF VOICE TRANSMITTERS	TUBE OUTGASSING CAUSES ARCING	TUBE SELECTION
	PRESSURE LOSS CAUSES ARCING	HERMETIC SEAL
C-BAND BEACON	MAGNETRON OUTGASSING CAUSES ARCING	TUBE SELECTION
	PRESSURE LOSS CAUSES ARCING	HERMETIC SEAL
S-BAND BEACON	TUBE OUTGASSING CAUSES ARCING	TUBE SELECTION
	PRESSURE LOSS CAUSES ARCING	HERMETIC SEAL
TRIPLEXER, QUADRI-PLEXER	PRESSURE LOSS CAUSES ARCING	HERMETIC SEAL
ANTENNAS	DIELECTRIC BREAKDOWN CAUSES ARCING	SELECT BETTER DIELECTRIC
SWITCHES	COLD WELDING OF CONTACTS	HERMETIC SEAL OR CABIN PRESSURIZATION

FIGURE 9.2-3

9.2.3 (Continued)

feels that for six to eight months of space storage the environment should be controlled between -20° and +130°F to prevent any degradation in performance.

Extremely limited data is available on the effects of radiation on propellants. However, this data indicates that no breakdown in N_2O_4 will occur when exposed to radiation. No data is available on MMH, but decomposition of a few percent has been experienced with UDMH under radiation. These decomposition products, primarily amines, are hypergolic with N_2O_4 and, consequently, no deleterious effects on propellant ignition or performance is anticipated. In addition, the tank walls will prevent or attenuate radiation effects on the propellants.

The ablative cooled Thrust Chamber Assemblies (TCA's) will experience some out-gassing in a vacuum environment. Ablative samples exposed to a vacuum environment for periods of up to two weeks have shown some loss of moisture and entrained gases. The TCA manufacturer feels that these effects do not shorten the life, and that the materials are satisfactory for a six to eight-month mission.

Exposure of Gemini type valves to pressures of 1×10^{-8} torr for 24 hours indicated no effects on opening or closing characteristics or on leakage rates. No test data is available for six months storage in space. However, the use of hard seats and dissimilar materials that are insoluble is expected to eliminate cold welds.

9.2.4 Environmental Control System - Contact with the Environmental Control System (ECS) manufacturer indicated that no studies or test data are available on the effects of prolonged exposure to the space environment. An investigation of the system was therefore made to determine the possible effects of standby and possible techniques for alleviating these effects.

Cold welding of metals in unpressurized areas is a potential standby hazard. Cold welding of metal to metal contact areas could occur in numerous valves.

9.2.4 (Continued)

These areas include seats, shafts, and sliding surfaces within the valve. It is anticipated that the cabin will be maintained at a pressure of 0.1 psia and at a temperature of 50°F during standby which should eliminate this possible problem with components located in this area. In unpressurized areas, cold welding can be prevented, or adverse effects reduced, by use of a dry lubricant or insoluble metals.

A permanent set of elastomeric valve seats may occur when valves are maintained in the closed position during standby resulting in system leakage and possible system malfunction during Ferry return. Elastomeric diaphragms in various components are also affected. These diaphragms are maintained in a compressed or expanded position for long periods of time and permanent set or degradation may occur. Upon activation of the system after standby, these diaphragms may rupture, leak or become inoperable, resulting in system failure. Proper selection of elastomeric materials used for valve seats and diaphragms is essential to insure against permanent set or degradation.

Water produced by the fuel cells during catch-up and rendezvous may freeze during standby and could result in ruptured water lines and tanks or inoperable valves. Electric heaters may be necessary to alleviate this condition.

Various system components contain static O-ring seals. Further investigation of seal material is desirable to determine degradation effects which may occur.

9.2.5 Pyrotechnic Equipment - Explosive devices are utilized to perform several functions in the Ferry Spacecraft after separation from the MORL. The explosive mixture is subject to chemical deterioration from the space environment, and the mechanical portions of the gas actuated devices are subject to cold welding and seal deterioration.

Chemical changes which will deactivate (dud) or reduce the performance of explosive devices can result from exposure to extreme temperature, radiation, and vacuum. Therefore, explosives should be hermetically sealed. Based on avail-

9.2.5 (Continued)

able data, low explosives used to operate mechanical actuators are more prone to deterioration. The high explosives are less affected and are more likely to function satisfactorily during or after exposure to extreme environment conditions.

Temperature - Thermal analyses of the Ferry Spacecraft have shown that low temperatures occur during standby in the areas containing pyrotechnic devices. The types of devices and their thermal limitations are summarized in Table 9.2-1. Although most devices are qualified to operate at temperatures near the minimum anticipated level, there is no data available on prolonged exposure to temperatures as low as -100°F. No problems are anticipated with high explosives at lower temperatures since launch vehicle destruct systems having flexible linear shaped charges have been tested successfully at the temperature of liquid hydrogen (-423°F). However, tests will be needed to verify the performance of any device having an anticipated temperature below the presently qualified value.

TABLE 9.2-1
TEMPERATURE LIMITATIONS – TYPICAL PYROTECHNIC DEVICES

TYPICAL DEVICES	TYPE	LOCATION	TEMPERATURE LIMITS – DEGREES F		ANTICIPATED MIN. TEMP. DURING STANDBY –°F
			STORAGE	OPERATING	
LINEAR SHAPED CHARGE TO SEVER STRUCTURE	HIGH EXPLOSIVE	OUTSIDE SURFACE OF ADAPTER	−100	−100	−103
WIRE BUNDLE GUILLOTINE	LOW EXPLOSIVE	INSIDE ADAPTER NEAR SKIN		−100	−56
MILD DETONATING FUSE TO SEVER RENDEZVOUS AND RECOVERY SECTION ATTACHMENT	HIGH EXPLOSIVE	FORWARD FACE OF R.C.S. PACKAGE	−100	−60	−150*
SWITCH TO DEAD FACE WIRE BUNDLES	LOW EXPLOSIVE	RE-ENTRY MODULE EQUIPMENT BAY	−65	−65	−65
TUBING CUTTER/SEALER	LOW EXPLOSIVE	INSIDE ADAPTER NEAR SKIN	−100	−100	−56
HATCH ACTUATOR	LOW EXPLOSIVE	CREW COMPARTMENT	−65	−65	−65
EJECTION SEAT CATAPULT	LOW EXPLOSIVE	CREW COMPARTMENT	−65	−65	−65

*OPERATES AFTER RE-ENTRY WHEN SECTION HAS WARMED UP.

9.2.5 (Continued)

Radiation - Available data on the effect of gamma radiation on explosives is extremely limited. However, this data indicates deterioration occurs at fairly high dose levels. On the basis of this data (Reference 9.2-1) no major problems are anticipated, although some modifications may be required in propellant composition. For example, in the prime explosives, lead styphinate appears far less susceptible to damage from radiation than any of the azides.

A test program to determine the limitations of all Gemini pyrotechnic mixtures is needed to verify radiation resistance.

Vacuum - Chemical changes of explosive materials in a vacuum occur due to outgassing of certain volatile constituents of the mixture. Gemini pyrotechnic devices, are designed for 14 days in space and explosives are hermetically sealed. Therefore, no problems due to vacuum are anticipated.

9.2.6 Landing Systems

Parachute - Main and drogue parachutes are subject to deterioration by thermal and high-energy particle radiation. One swivel fitting is subject to cold-welding. Parachute lines and bridles, however, should be unaffected. Pressurization to 0.1 psia of the drogue and main parachute compartments may be needed to alleviate the effects of long exposure to hard vacuum.

Parasail and Paraglider - The effect of the space environment on the parasail or paraglider is similar to that of the parachute system. In addition, the potting compounds, used in suspension line guide channels, and the oil-filled landing gear struts may be degraded by temperature extremes and radiation. Cable reels, drums, pulleys and swivel fittings are subject to cold welding and the inflation nitrogen bottles to leakage. Leakage can be combatted by careful seal design and by oversizing supply tanks.

9.2.7 Mechanical Systems - Available data indicates mechanical systems such as hatches, hinges, latches and ejection seat slide blocks, are generally not

9.2.7 (Continued)

limited by the environmental hazards of space. Phenomena such as cold welding can be avoided through the use of proper lubricants, hard finishes, and insoluble dissimilar materials.

9.2.8 Materials - Laboratory tests have shown that although metals and alloys are generally quite stable in the high vacuum and solar ultraviolet environment of space, cold welding can occur between surfaces which are in contact for long periods. Most inorganic materials are not affected by high vacuums, unless accompanied by moderately high temperatures. Some plastics, such as nylon, acrylic, or neoprene, decompose readily in a vacuum while others, such as butadiene, styrene, isoprene, or natural rubber, exhibit low degradation rates. Outgassing and evaporation of lubricants and certain other materials may be a problem. Certain glasses lose optical properties in a vacuum ultraviolet environment, and some thermal radiation coatings encounter property changes. For example, the solar absorptivity/infrared emissivity ratio on the aft cover of the adapter decreases in a vacuum environment. Careful material selection, and appropriate testing will be needed to assure satisfactory characteristics.

9.3 Protective Measures - Protection for systems which are subject to deterioration in the space environment can be provided by several methods. These include spacecraft temperature control by either active or passive means, maintenance of a pressurized atmosphere, and radiation and meteoroid shielding.

Systems such as rendezvous radar are not used after completion of mooring, and therefore do not require protection during standby. Other systems, which might be used, have inherent features which make them difficult to protect during long exposure in space. In some of these cases, it is simpler to use an alternate system after standby rather than protect a particular system. These cases are summarized in Table 9.3-1.

TABLE 9.3-1
SYSTEMS ELIMINATED FROM POST STANDBY OPERATION

FUNCTION	SYSTEM THAT WOULD NORMALLY BE USED BUT ELIMINATED DUE TO STANDBY PROBLEMS	ALTERNATE USED IN LIEU OF ELIMINATED SYSTEM
STATION SEPARATION	ORBIT ATTITUDE AND MANEUVER SYSTEM (OAMS)	TWO SOLID ROCKETS
ATTITUDE CONTROL & RETRO POSITIONING	OAMS	RE-ENTRY CONTROL SYSTEM
ELECTRIC POWER	FUEL CELLS	AUTO-ACTIVATED BATTERIES
SPACECRAFT COOLING	SPACE RADIATOR	WATER EVAPORATOR
BREATHING OXYGEN SUPPLY	SUPERCRITICAL OXYGEN STORAGE SYSTEM	ORBIT OXYGEN ADDED TO GASEOUS SECONDARY OXYGEN SUPPLY

9.3.1 Thermal Protection - Thermal protection is necessary during standby for numerous components.

Ferry Spacecraft - Electrical heaters in the spacecraft, using power from the MORL, provide thermal protection with minimum change to Gemini. When two Ferry Spacecraft and one Unmanned Supply Spacecraft are moored on the station, the maximum heater power required is approximately 500 watts.

Adding electrical resistance heaters to the Gemini temperature control system

9.3.1 (Continued)

is a simple method of maintaining allowable temperatures during standby. The heater power required for an orientation resulting in the lowest adapter temperatures as described in Section 9.1 was determined for large components only, since raising the temperature of these items also raises the temperature of small components. A summary of Ferry component temperatures and heater power requirements is presented in Table 9.3-2. Not all spacecraft require this amount of heat simultaneously, however, since some are moored in warmer orientations. Some considerations leading to the approach of providing heaters are:

A. A nose moored spacecraft with the aft cover 180° from the sun's rays requires almost the same heater power in the adapter (110 watts) as in the re-entry module. Mooring the spacecraft with the aft cover towards the sun requires redesign of the Gemini temperature control system, including a new coating on the aft cover to lower the maximum standby temperatures. If this coating is changed, catch-up orbit temperatures become too cold with the new coating and heaters are needed during this mission phase.

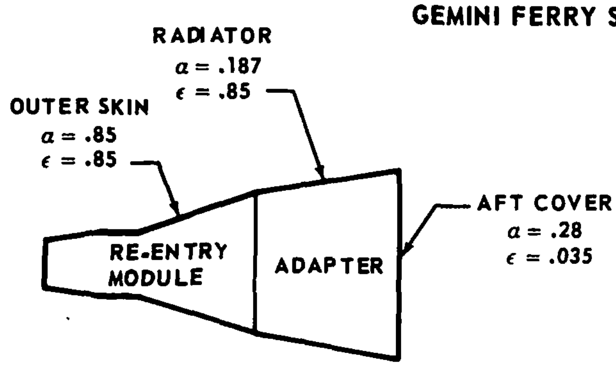
B. Spacecraft which are side moored with their longitudinal axes normal to the sun's rays need approximately those heaters shown in Table 9.3-1. A spacecraft which is side moored on the sun side of the MORL does not appear to need heaters for adapter equipment, since the heat absorbed by the aft cover appears to be sufficient to maintain desired temperatures.

C. Spacecraft which are aft moored to the station have heater requirements similar to the nose moored spacecraft. No redesign of the basic Gemini temperature control coatings is required.

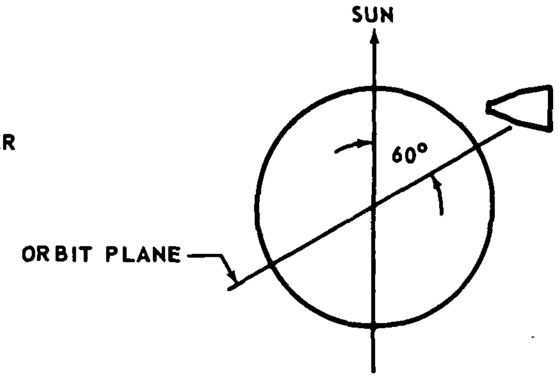
Ferry/Supply Spacecraft - The Ferry portion of these spacecraft needs the same type of thermal protection as the Ferry spacecraft discussed above.

The cargo sections require coatings, heaters, and appropriate tankage arrangements for thermal protection. Approximately 370 watts of heater power is necessary

TABLE 9.3–2
STANDBY THERMAL PROTECTION
GEMINI FERRY SPACECRAFT



NOTE: α – SOLAR ABSORPTIVITY
ϵ – INFRARED EMISSIVITY



ASSUMED SPACECRAFT ORIENTATION

COMPONENT	ALLOWABLE STANDBY TEMPERATURE °F	MINIMUM TEMPERATURE WITHOUT HEATING °F	MINIMUM TEMPERATURE WITH HEATING °F	HEATER POWER (WATTS)	COMMENTS
ADAPTER SECTION					
ECS PUMP PACKAGE	32	–55	32	110	HEAT SUPPLIED HERE IS SUFFICIENT TO WARM SURROUNDING COMPONENTS TO –8°F
RETROGRADE ROCKETS	–20	–55	–8	0	
BATTERIES	–50	–55	–8	0	
MISCELLANEOUS (INCLUDES SEPARATION ROCKETS, FUEL & OXIDIZER LINES, VALVES, ETC.)	20	–50 to –100	20	10	
RE-ENTRY MODULE					
CABIN					
EQUIPMENT BAY (RIGHT)	20	–30 (1)	20	17(1)	HEATER SIZE = 32 WATTS
EQUIPMENT BAY (BOTTOM)	20	–65	20	5	
EQUIPMENT BAY (LEFT)	20	–65 (1)	20	32 (1)	
ECS PACKAGE (BOTTOM)	32	–65	32	6	
HATCH AREA (INCLUDES LATCH & HINGES)	25	–95	–58	0	17 WATTS APPLIED JUST PRIOR TO FERRY INGRESS
RCS SECTION					
PROPELLANT TANKS	20	–50	20	40	
VALVE PACKAGES	20	–35	20	2	
LINES TO THRUSTERS	20	–35	–30	0	32 WATTS APPLIED JUST PRIOR TO FERRY INGRESS
SOLENOID VALVES	20	–50	–45	0	
TOTAL POWER REQUIRED FROM STATION				222	49 WATTS REQUIRED JUST PRIOR TO FERRY INGRESS

(1) TEMPERATURE AND HEATER POWER REQUIREMENTS ALTERNATE WITH SPACECRAFT ORIENTATION.

9.3.1 (Continued)

to maintain a temperature of 20°F in the pressurized section. This level is needed because the high surface emissivity (0.9) used to maintain the magnesium external structure at allowable temperatures during launch results in low temperatures in orbit.

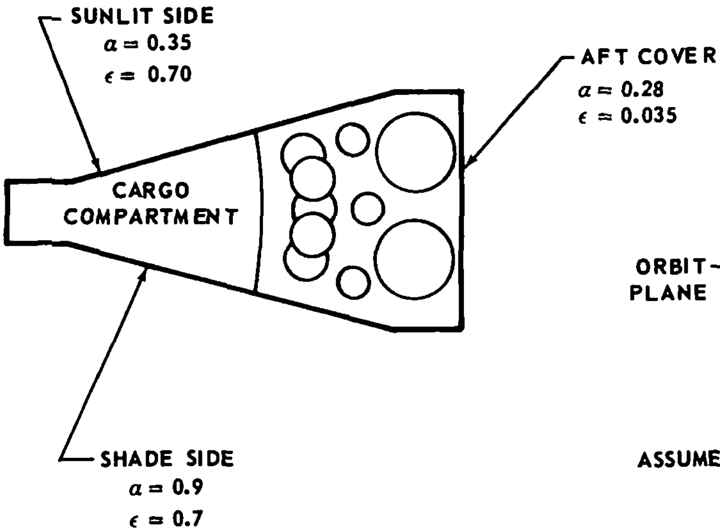
Although the aft cover on the Ferry section is not exposed during catch-up orbits, the temperatures are above the allowable minimums.

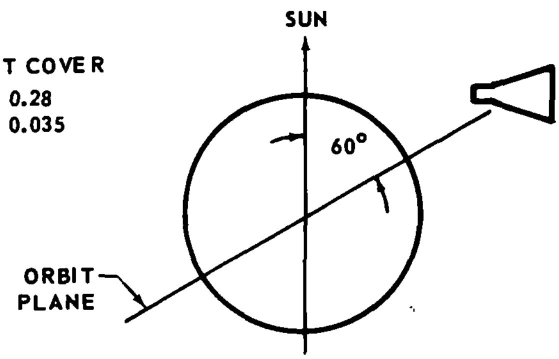
Unmanned Supply Spacecraft - Surface coatings for passive thermal protection of the Specific Design Unmanned Supply Spacecraft were investigated. For the spacecraft orientation resulting in the lowest adapter temperatures of the Ferry (Section 9.1), one side of the spacecraft receives direct solar radiation. The opposite side receives earth emitted and earth reflected radiation only. To maintain the desired temperature environment in the Supply Spacecraft it is necessary to use different coatings over the external surface. An absorptivity/emissivity ratio of 0.5 is used for the sunlit surface, to maintain the desired temperature environment, as shown in Table 9.3-3. For the shaded surface, a large ratio of solar absorptivity to infrared emissivity is desired. Launch heating restricts the minimum value of surface emissivity to 0.7, therefore, the highest practical ratio obtainable is about 1.30. Analyses show that passive control with this coating is inadequate to maintain the desired minimum temperature on the shaded side of the spacecraft. To obtain a minimum temperature of 20°F in the pressurized cargo compartment, blanket type heaters are used since the heat is required over a large surface area. The power requirement is 121 watts.

The oxidizer and pressurant tanks are located on the sunlit side of the spacecraft and the fuel tanks on the shaded side. Analysis shows heaters are not needed for the tanks located in the unpressurized section, since temperature control is accomplished with the aft cover coated the same as on Gemini.

TABLE 9.3-3

STANDBY THERMAL PROTECTION
UNMANNED SUPPLY SPACECRAFT
SPECIFIC DESIGN





ASSUMED SPACECRAFT ORIENTATION

NOTE: α – SOLAR ABSORPTIVITY
ϵ – INFRARED EMISSIVITY

COMPONENT	ALLOWABLE STANDBY TEMPERATURE – °F	TEMPERATURE EXTREME WITHOUT HEATING – °F	TEMPERATURE EXTREME WITH HEATING – °F	HEATER POWER (WATTS)	COMMENTS
EXTERNAL SKIN SUNLIT SIDE	+200 MAX.	125 MAX.	125 MAX.	0	
CARGO ON SUNLIT SIDE	+100 MAX.	70 MAX.	70 MAX.	0	
EXTERNAL SKIN SHADED SIDE	–150 MIN.	–129 MIN.	–83 MIN.	120	BLANKET HEATER
CARGO ON SHADED SIDE	20 MIN.	–42 MIN.	20 MIN.		
OXIDIZER AND PRESSURANT TANKS	20 MIN.	29 MIN.	29 MIN.	0	
FUEL TANKS	–45 MIN.	–53 MIN.	–53 MIN.	0	LOW TEMPERATURE WILL BE TOLERATED
LINES, VALVES AND NOZZLES	20 MIN.	-50 TO –100 MIN.	20 MIN.	20	

9.3.1 (Continued)

A number of smaller components, such as the separation rockets, orbit maintenance thrust chambers, and various valves and lines need heater power. Based on Gemini studies, 20 watts is estimated to be sufficient to maintain a suitable temperature environment.

The Stripped Gemini Unmanned Supply Spacecraft has a radiator on part of the external adapter surface. This surface uses a low solar absorptivity to infrared emissivity ratio coating to obtain maximum performance of the radiator, therefore low surface temperatures result during standby. With the oxidizer and pressurant tanks placed on the sunlit side of the adapter, a small amount of heater power is still necessary. The pressurized storage compartment is temperature controlled with blanket electric heaters, requiring a total power of 300 watts.

During the catch-up and rendezvous period no temperatures below the minimum allowable are experienced for either version.

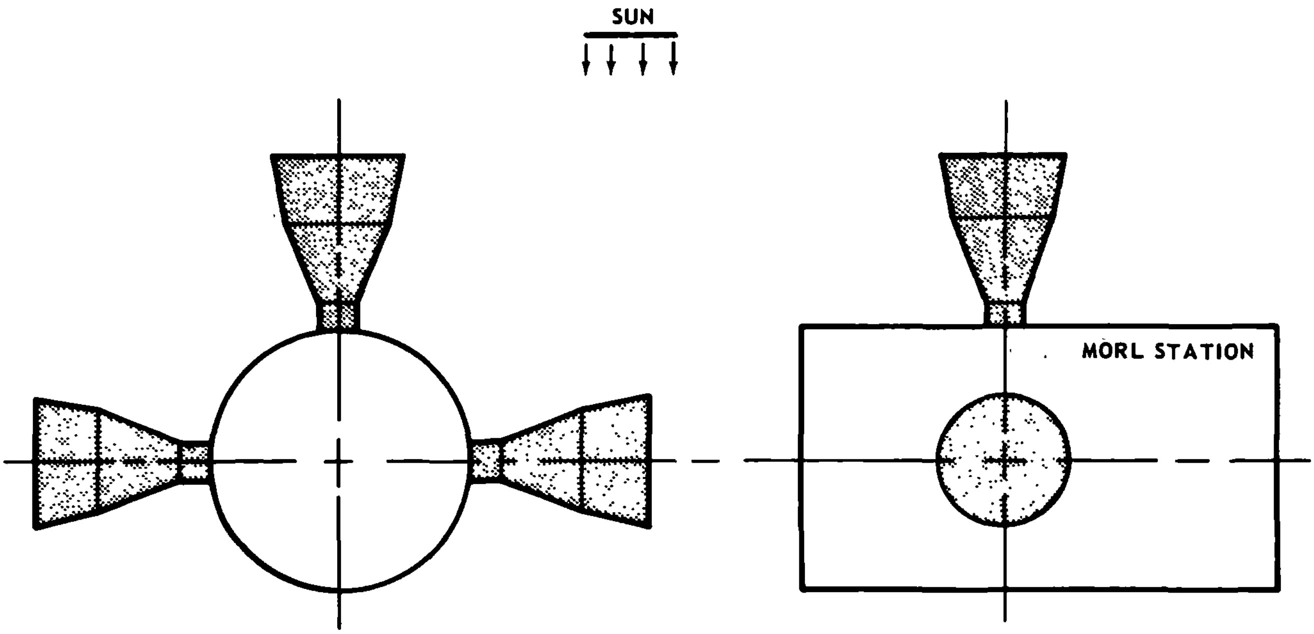
Alternate Thermal Protection Methods - Several methods of providing passive thermal protection for the spacecraft or of providing heat with the use of the coolant loop in Gemini were investigated.

Completely passive temperature control of equipment located in the Ferry adapter appears feasible when the spacecraft are angle moored to the MORL station as shown in Figure 9.3-1. In the angled concept the angle between the sun's rays and the spacecraft longitudinal axis is approximately 70°, which provides only partial shadowing of the concave Gemini aft cover. This raises the temperature sufficiently in the adapter to eliminate the need for heaters. Electrical heaters are still needed, however, to maintain temperatures in the re-entry module.

Passive temperature control is also feasible if external covers, having appropriate coatings are mechanically applied in space. These covers, or bags, can be attached with clips or zippers, or tied over the spacecraft. Encapsulation

COMPARISON OF PRESENT AND ANGLED MOORING CONCEPTS

PRESENT MOORING CONCEPT



ANGLED MOORING CONCEPT

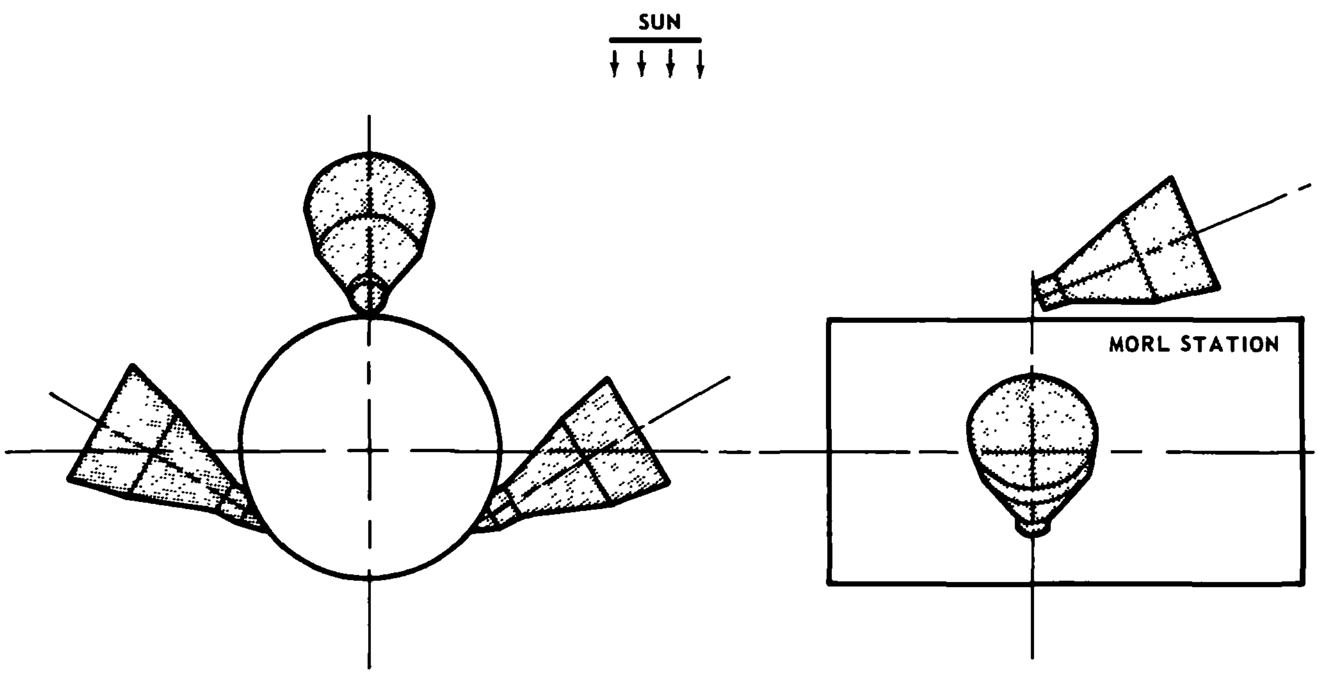


FIGURE 9.3-1

9.3.1 (Continued)

concepts are discussed in Section 6.7.

The Ferry coolant loop could also supply heat to the spacecraft by connecting a coolant loop from MORL to the Ferry radiator circuit. This system provides standby thermal protection and also provides a greater effective MORL radiator area. Initial investigation of this system determined it to be feasible, but the reliability of the Ferry coolant system pumps is greatly reduced since at least one pump must operate continuously during standby. New pumps with much higher reliabilities would have to be developed to make this method of providing standby thermal protection attractive.

9.3.2 Pressurization - Possible cold welding of metals in contact and outgassing of certain materials are hazards of prolonged exposure to hard vacuum. Maintaining a relatively low pressure in the spacecraft pressure vessel during standby would alleviate these effects for the equipment located in this area. A constant bleed orifice in a line from MORL will be used to maintain this pressure. A total of 35 pounds of oxygen at design leakage rates is required to maintain 0.1 psia in all Ferry spacecraft moored at the MORL for a combined period of 690 days.

Components requiring significantly higher pressures (e.g., the inertial platform) and items in the adapter will be independently pressurized and sealed.

Pressurization for the cargo section of the supply modules is provided by the MORL, as discussed in Section 7.7.

9.3.3 Radiation Protection - Protection against radiation damage can be achieved by the addition of shielding to the spacecraft or components. However, analysis of known equipment sensitivities to radiation has not shown a positive need for any added shielding for standby protection even under the severe July 1962 environments.

9.3.4 Meteoroid Protection - Multi-walled structure is the lightest approach to providing a given level of protection against meteoroid penetration. No analysis of added protection has been made, however, for the Ferry Spacecraft during standby because of the inherent low penetration probability shown in Figure 9.1-3.

9.4 Minimum Systems Required for Safe Return to Earth - Although the primary consideration in providing standby protection is to permit normal operation of the Ferry Spacecraft after standby, exceptional attention should be given to those operations and systems which are essential to insure safe re-entry and landing. Safe return is dependent primarily on the availability of electric power and breathing oxygen and on the operation of several pyrotechnic devices and the Re-entry Control System. The necessary operations and the associated required systems, along with the amount of back-up, are shown in Table 9.4-1.

Probability of Safe Return with Minimum Systems - An estimate of the probability of a successful return from a 250 nautical mile orbit is based on the following assumptions:

A. A successful return means a safe re-entry and landing at the first possible landing point available, using the minimum required systems.

B. The retrograde and re-entry phase is one hour in duration.

C. The systems required for a successful return are as shown in Figure 9.4-1.

With the environmental protection that is provided for the minimum systems it is estimated that reliability degradation during standby will be negligible.

The probabilities of success for each required system, together with the total probability of a successful return are presented in Table 9.4-2. The system with the greatest contribution to the total probability of failure is the Re-entry Control System. The slight degradation in success probability, compared to Gemini, is due primarily to the longer re-entry time and the use of five of six retro-rockets instead of three of four.

TABLE 9.4–1
MINIMUM SYSTEMS FOR SAFE RETURN TO EARTH

REQUIREMENT	PRIMARY SYSTEM	BACKUP CAPABILITY
ELECTRIC POWER	AUTO-ACTIVATED BATTERIES 7 ON MAIN BUS 3 ON SQUIB/CONTROL BUS	FOR MINIMUM SYSTEM OPERATION: ONLY 2 SQUIB/CONTROL BATTERIES REQUIRED
ASTRONAUT BREATHING AND SUIT PRESSURIZATION	TWO OXYGEN BOTTLES	ONE OXYGEN BOTTLE SUFFICIENT FOR BOTH ASTRONAUTS
RELEASE MOORING LATCHES	MECHANICAL RELEASE ON MORL	PYROTECHNIC RELEASE ON GEMINI
SEPARATE FROM STATION	TWO SOLID ROCKETS	ONE ROCKET PROVIDES SUFFICIENT SEPARATION IMPULSE
ATTITUDE INDICATION	INERTIAL REFERENCE SYSTEM	VISUAL REFERENCE THROUGH WINDOW
POSITION TO RETRO ATTITUDE AND HOLD	RE-ENTRY CONTROL SYSTEM	TWO COMPLETELY SEPARATE SYSTEMS INSTALLED – ONE REQUIRED
JETTISON EQUIPMENT SECTION	PYROTECHNIC SHAPED CHARGE, WIRE BUNDLE GUILLOTINES, AND TUBE CUTTERS	ALL DEVICES INSTALLED IN PAIRS FOR 100% REDUNDANCY
RETROGRADE	5 SOLID ROCKETS	6 ROCKETS INSTALLED TO PROVIDE 1 BACKUP
JETTISON RETROGRADE SECTION	PYROTECHNIC SHAPED CHARGE	DUAL INSTALLATION FOR 100% REDUNDANCY
RE-ENTRY CONTROL	RE-ENTRY CONTROL SYSTEM	TWO COMPLETELY SEPARATE SYSTEMS INSTALLED – ONE REQUIRED
RECOVERY AND LANDING	PARACHUTE, PARASAIL OR PARAGLIDER	EJECTION SEAT SYSTEM

TABLE 9.4–2
MINIMUM SYSTEMS RELIABILITY

REQUIRED SYSTEMS	P_S(FERRY MISSION RETURN FROM 250 NA.MI.)
RELEASE FROM MORL	1.00000
RE-ENTRY CONTROL SYSTEM	.99973
RETRO-ROCKETS AND SEPARATION ROCKETS	.99999
GUIDANCE AND CONTROL ELECTRONICS	.99990
ELECTRICAL POWER	.99993
ENVIRONMENTAL CONTROL SYSTEM	.99999
LANDING SYSTEM *	
a. DROGUE DEPLOY AND MORTAR	.99800
b. MAIN CHUTE (SEQUENCE)	.99758
c. EJECTION (SEQUENCE)	.98605
PYROTECHNICS	
a. EQUIPMENT SECTION–RETRO PACK SEPARATION	.99999
b. RETRO SEAT JETTISON	.99999
c. RETRO AND RE-ENTRY SEAT JETTISON	.99999
d. HORIZON SCANNER FAIRING JETTISON	.99980
e. HORIZON SCANNER JETTISON	.99980
TOTAL PROBABILITY OF SUCCESSFUL RETURN	.99907

* SEAT EJECTION SEQUENCE IS A BACKUP TO THE MAIN CHUTE SEQUENCE.

9.5 Monitor and Checkout

9.5.1 Ferry Shutdown - When the Ferry has been docked at the MORL, subsystems, except the Environmental Control System (ECS), Orbit Attitude and Maneuver System (OAMS), communications and electrical power, are deactivated. One astronaut transfers to the MORL to assist the second astronaut in the shutdown sequence and in the transfer of system monitoring to the MORL. The second astronaut leaves the Ferry, after shutdown of remaining systems closes and seals the hatches and enters the MORL.

The shutdown and stand-by preparation sequence includes these major items:

Docking - umbilical connection completed.

Attitude Control and Maneuver Electronics (ACME) logic-off.

Rate Gyros - off.

Horizon Scanners - off.

Rendezvous Radar - off.

Computer, Inertial Measurement Unit and Attitude Display Group - off.

C and S-Band Beacons - off.

Digital Command System and Time Reference System - off.

Acquisition Aid Beacon - off.

Prepare for egress, switch to closed suit circuit mode for oxygen supply and pressure.

Cabin heater, cooling system and O_2 supply - off.

Open hatches.

Co-astronaut leaves Ferry and enters MORL.

MORL power to Ferry - on.

Main, squib and control buses - off, Fuel cells - off.

Vent fuel cell hydrogen and LOX systems. Water transfer line - closed.

Switch Telemetry System and Digital Command System from antenna to umbilical coax.

9.5.1 (Continued)

Vent OAMS propellant and purge system to space.

MORL to cabin oxygen bleed - on.

Standby heaters - on.

Coolant system - off.

Spacecraft Monitoring System - on.

Commander leaves Ferry and enters MORL.

Cabin lights - off. (Control from MORL).

Voice communications - off. (Control from MORL).

9.5.2 Standby - During the standby period the crew in the MORL will assess the condition of the moored Ferry Spacecraft to be assured of its continuous operational capability for a return flight. The feasibility of various methods of monitor and checkout of the necessary systems was evaluated based on the following considerations:

A. There does not appear to be any major problem in assuring reliable operation of ferry equipment and systems after long inoperative standby in space. Equipment which cannot be re-designed for prolonged storage or passively protected against deterioration in space can be actively protected by heaters, continuous pressurization, and, in some cases, by periodic operation to reverse a deteriorating condition. Equipment will be thoroughly ground tested to provide assurance of its capability of operating after exposure to the space environment.

B. Minimum systems required for safe return to earth (Section 9.4) have a high degree of redundancy and reliability and will receive special protection during standby.

C. Active maintenance in space involves extravehicular operations. In the MORL time period such techniques will not have been fully developed and, therefore, are not considered as part of the operations plan.

9.5.2 (Continued)

D. It is impractical to exercise equipment to check for proper operation since there is no assurance that it will again operate properly the next time it is turned on. Also, this type of random operation may cause a failure. For example, if a valve, which is normally closed and properly sealed, is opened to check its operation, it may not reseat properly and excessive leakage could occur.

Monitor and Checkout Methods - It is assumed that extensive maintenance while moored is impractical for the mooring positions considered. Therefore, the monitor and checkout functions are limited to the following:

A. Status of Controlled Environment - Since some equipment will be protected by controlling the temperature, pressure and radiation environment, variations from allowable limits must be detected in order to permit corrective action.

B. Status of Stored Expendables - Breathing oxygen and reaction control system propellant and pressurant are stored in containers which are subject to leakage. Quantities will be monitored to detect excessive leakage and permit return of the Ferry to earth before these expendables are depleted below a safe level.

C. Continuity of Squib and Control Electrical Circuits - Most of the minimum systems required for safe return to earth are electrically activated with dual circuitry. Monitoring of circuit resistance and continuity will indicate any deterioration.

Indications from any of these data that a critical system is inoperative would entail repairs, when feasible, or rescue of the crew.

Measurements - Data described in Section 9.5.1 will be collected with the following measurements on board the Ferry.

A. Temperatures at various points throughout the spacecraft.

B. Pressure in compartments requiring pressure control.

C. Accumulated radiation dose at various points.

9.5.2 (Continued)

D. Temperature and pressure of stored gases from which the remaining quantity can be determined.

E. Quantity of stored liquids. (Method to be determined.)

Monitor and Checkout System - Minor modification of Gemini equipment and inclusion of additional sensors will provide a checkout and monitor capability for the Ferry Spacecraft. The basic elements are the Telemetry System, associated sensors or test points, and the Digital Command System. Additional sensors for measurements requiring continuous monitor will be directly connected to the MORL display panel through the docking umbilical. Systems that are considered critical are listed in Table 9.4-1.

Temperatures and pressures at selected key locations will be measured and the status presented continuously at the MORL checkout and monitor station. These displays will be reviewed several times a day as part of the regular crew duty cycle. Out-of-tolerance indications will dictate a full standby checkout procedure.

Other measurements will be checked on a periodic schedule measured in weeks. Power will be applied and telemetry equipment will be remotely activated from the MORL checkout station through docking umbilical connections. The measurements will be multiplexed by the telemetry equipment for transmission to the MORL checkout station displays. During the Ferry standby preparations, the telemetry transmitter will have been switched from the antenna to the umbilical co-axial cable. Figure 9.5-1 presents a block diagram of the checkout and monitor system.

The predeparture checkout is supplemented by using the Digital Command System to activate equipment with long warm-up times, such as the inertial platform. This system can also be used to turn on systems which need a full checkout.

9.5.3 Pre-departure Checkout - The pre-departure checkout informs the crew

CHECKOUT AND MONITOR DURING STANDBY

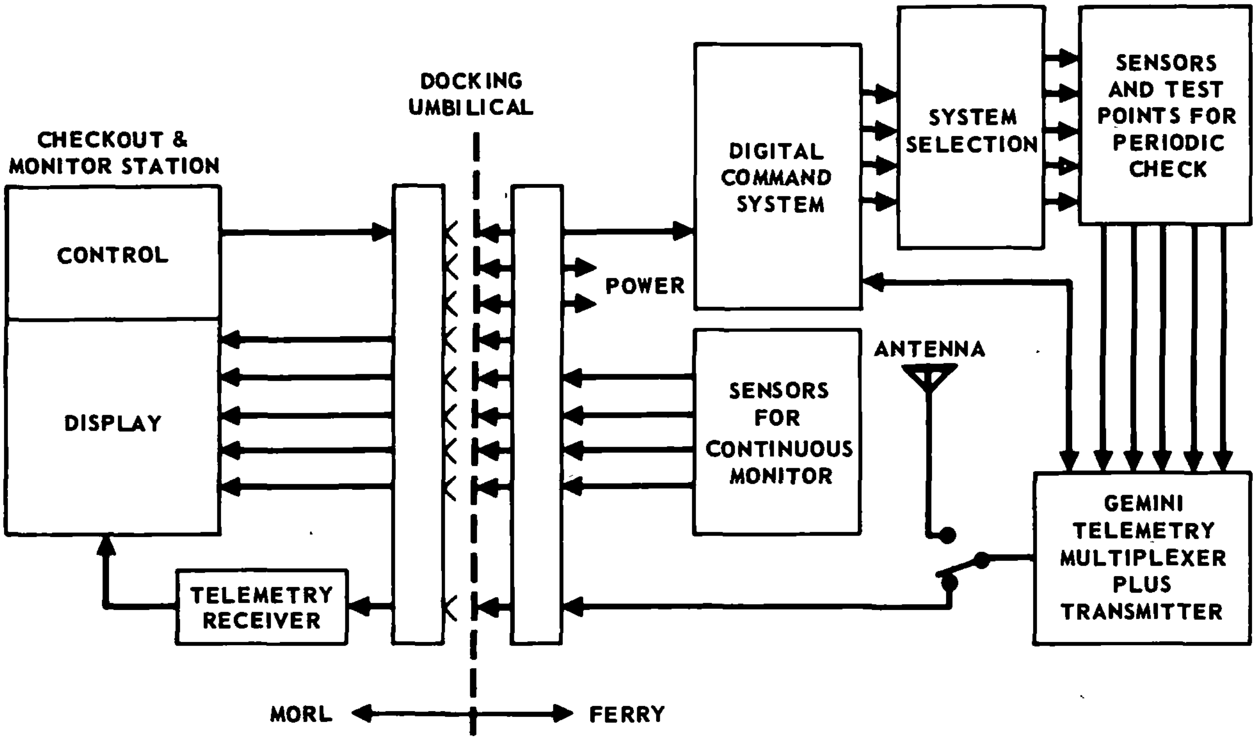


FIGURE 9.5-1

of the operability of each subsystem allowing them to compensate for any equipment failures, or return to standby for emergency repair or request for rescue. The procedures consist mainly of operational checks by the astronauts in the Ferry in conjunction with ground and MORL personnel, where required, to determine go/no-go status.

Additional equipment is not required on Gemini for these tests. However, several controls, not presently available to the astronaut, such as the inertial platform trim potentiometers, are brought out to the control panel. The voice transceivers, telemetry transmitters, and the digital command system are checked by operating each Ferry system in conjunction with its MORL counterpart via the umbilical RF coax link. This procedure is repeated with systems switched to their antennas to check for antenna breakdown. Radiated signal strengths are monitored in MORL, thus eliminating dependence on ground station cooperation.

9.5.3 (Continued)

The pre-departure checkout sequence includes these major items:

Activate batteries (control from MORL).

Inertial measurement unit heaters - on (control from MORL).

Voice communications - on (control from MORL).

Cabin lights - on (control from MORL).

Commander leaves MORL and enters Ferry.

Spacecraft monitoring system - off.

Coolant system - on.

Standby heaters - off.

MORL to cabin oxygen bleed - off.

Main squib and control busses - on.

MORL power to Ferry - off.

Co-pilot leaves MORL and enters Ferry.

Seal hatches and pressurize cabin.

Cabin heater, cooling system and O_2 supply - on.

Astronauts switch to Ferry ECS.

Acquisition Aid Beacon - on.

C and S - Band Beacons - ground command.

Digital command system - on and test.

Time reference system - on and reset.

Computer, inertial measurement unit and attitude display group - on - start computer diagnostic routine.

Rate gyros - on. Check spin motor rotation detector.

Trim platform gyro drift bias currents and pulse rebalance electronics bias currents with cockpit controls.

Apply torquing current to rate gyros and observe response on attitude display group.

9.5.3 (Continued)

Horizon scanners - on.

Attitude control and maneuvering logic - on.

Static fire all unobstructed thrusters.

Release latching mechanism.

Fire separation rockets.

10. RELIABILITY

The mission reliabilities of each of the spacecraft studied are presented together with the assumptions and ground rules upon which the estimates are based. The system estimates presented in Section 7 are combined with the overall mission diagrams and time profiles to give mission reliability estimates.

10.1 Mission Definitions - The reliability diagrams presented in Figure 10.1-1 divide each mission into phases and establish those spacecraft systems which must function during each phase.

The mission phases are:

Launch - Launch begins with space vehicle lift-off and ends with launch vehicle/spacecraft separation.

Self-Inject (Unmanned Supply only) - Self-inject begins with launch/vehicle spacecraft separation and ends with the signal for self-inject thrust termination.

Catch-up - The catch-up phase begins with launch vehicle/spacecraft separation and ends with the initiation of terminal guidance.

Terminal Guidance - The terminal guidance phase begins with radar and/or visual target (MORL) acquisition and ends at spacecraft docking.

Mooring - The mooring phase begins when a rigid connection has been achieved and ends when preparation for spacecraft standby is completed.

Orbital Standby - The orbital standby phase begins with standby preparation completion and ends with the signal to (1) activate the spacecraft systems in preparation for re-entry (Ferry) or (2) physically separate from the laboratory in preparation for next supply (supply module with multiple re-supply).

Separation and Parking - (Ferry) - The separation and parking phase begins with the signal to activate the Ferry Spacecraft in preparation for re-entry and ends with the signal for retrofire.

MISSION OVER-ALL RELIABILITY DIAGRAMS

I. FERRY SPACECRAFT

A. LAUNCH PHASE

→ LAUNCH VEHICLE – GUIDANCE & CONTROL SYSTEM* – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SCANNER FAIRING JETTISON – RADAR TRANSPONDER FAIRING JETTISON – LAUNCH VEHICLE/SPACECRAFT SEPARATION →

B. CATCH-UP PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM →

C. TERMINAL GUIDANCE PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATION – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SPACECRAFT DOCKING MECHANISM →

D. MOORING PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SPACECRAFT MOORING EQUIPMENT – MORL LABORATORY →

E. ORBITAL STANDBY PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SPACECRAFT MOORING EQUIPMENT – MORL LABORATORY →

F. SEPARATION & PARKING

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SPACECRAFT DOCKING & MOORING MECHANISM – EQUIPMENT SECTION/RETRO SECTION SEPARATION – MORL LABORATORY →

G. RETROGRADE & RE-ENTRY PHASE

→ RETROGRADE ROCKETS – RETRO SECTION JETTISON – SCANNER HEAD JETTISON – GUIDANCE & CONTROL SYSTEM – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM →

H. LANDING PHASE

→ ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – LANDING SYSTEM →

II. UNMANNED SUPPLY SPACECRAFT

A. LAUNCH PHASE

→ LAUNCH VEHICLE – GUIDANCE & CONTROL SYSTEM* – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – RADAR FAIRING JETTISON – SCANNER FAIRING JETTISON – LAUNCH VEHICLE/SPACECRAFT SEPARATION →

B. INJECTION PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SUPPLY STORAGE – INJECTION THRUST SYSTEM →

C. CATCH-UP PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SUPPLY STORAGE →

D. TERMINAL GUIDANCE PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SPACECRAFT DOCKING MECHANISM – SUPPLY STORAGE →

E. MOORING PHASE

→ GUIDANCE & CONTROL SYSTEM – COMMUNICATIONS – INSTRUMENTATION – ELECTRICAL POWER SYSTEM – ENVIRONMENTAL CONTROL SYSTEM – SPACECRAFT MOORING EQUIPMENT – SUPPLY STORAGE – MORL LABORATORY →

F. ORBITAL STANDBY PHASE

→ INSTRUMENTATION – PROPULSION – ENVIRONMENTAL CONTROL SYSTEM – SUPPLY STORAGE – SPACECRAFT MOORING EQUIPMENT – MORL LABORATORY – SUPPLY MODULE/MORL LABORATORY SEPARATION →

*THE PROPULSION SYSTEMS AS WELL AS THE RADAR SYSTEMS ARE INCLUDED UNDER THE HEADING OF GUIDANCE AND CONTROL.

FIGURE 10.1-1

10.1 (Continued)

Retrograde and Re-entry - (Ferry) - The retrograde and re-entry phase begins with the signal for retrofire and ends with drogue chute deployment.

Landing - The landing phase begins with drogue chute deployment and ends with spacecraft touchdown.

Duty cycles for the spacecraft systems were established for each spacecraft using mission event time profiles, which are presented in Figure 10.1-2 and are based on the following assumptions and ground rules.

MISSION EVENT-TIME PROFILES

PHASE	FERRY	FERRY SUPPLY	UNMANNED SUPPLY
BOOST	.10	.10	.10
INJECTION	N/A	N/A	.17
ORBIT			
CATCH-UP	19.90	19.90	19.90
TERMINAL GUIDANCE	2.60	2.60	2.60
MOORING	1.00	1.40	1.40
STAND-BY	–	–	–
SEPARATION & PARK	3.50	N/A	N/A
RETROGRADE & RE-ENTRY	1.00	N/A	N/A
LANDING	.40	N/A	N/A
TOTAL OPERATIVE TIME (HR.)	28.5	24.0	24.17

FIGURE 10.2-2

A. The MORL is in a 250 na. mi. circular orbit.

B. Radar acquisition occurs at a range of 250 na. mi.

C. The maximum time required for crew transfer is one hour.

D. The time required to enter the Ferry and separate the Ferry from the MORL is .5 hour. Two parking orbits are assumed prior to retrograde.

E. The supply mission is defined to begin at lift-off and end at spacecraft separation from the MORL. The Ferry and Ferry/Supply mission is defined to begin at lift-off and end at spacecraft touchdown.

10.1 (Continued)

F. The Ferry/Supply Spacecraft is separated at the MORL and the Ferry and Supply modules are moored separately. The Supply module either remains moored to the laboratory for the remainder of the MORL mission or is jettisoned when expended in preparation for the next supply spacecraft.

10.2 Mission Reliability Estimates - Mission estimates were computed for the Ferry, Ferry/Supply, Stripped Gemini and the Specific Design Supply Spacecraft. Mission reliability is defined as the probability that the crew and the spacecraft subsystems will perform all functions and operations necessary for achieving each mission objective.

A. The reliability of the crew is equal to unity.

B. The reliability of the ground equipment is equal to unity.

C. The reliability of required MORL equipment is equal to unity, unless specifically stated otherwise.

D. With the environmental protection that is provided, it is estimated that reliability degradation during standby will be negligible.

E. The failure rate estimates currently being used in Gemini analyses are applied. Failure rate estimates for the non-Gemini equipment are derived from McDonnell test data and current component failure rate publications.

F. System failures during the launch to mooring phase which necessitate the use of post-separation systems, which are not normally used prior to separation, constitutes a mission failure.

Mission Estimates - The mission estimates are presented in Table 10.1-1. The mission unreliability for each spacecraft is attributed primarily to the guidance and control system. This is due to the complexity of the electronics systems, the operational leakage failure rates for the propulsion propellant systems, and the high frequency of propellant valve actuation for the rendezvous mission. Each of the remaining systems are less severe in their effect on the total mission reliabilities.

TABLE 10.1-1
RELIABILITY MISSION ESTIMATES

SYSTEM OR FUNCTION	FERRY RELIABILITY (OPERATIVE PHASES – TOTAL MISSION)	PERCENT OF TOTAL MISSION UNRELIABILITY
GUIDANCE & CONTROL	.9677	74.22
ENVIRONMENTAL CONTROL	.9978	5.11
ELECTRICAL POWER	.9974	6.04
COMMUNICATION (VOICE & GROUND TRACKING)	.9998	.46
INSTRUMENTATION	.9998	.46
SEQUENTIALS & RETRO-ROCKETS	.9993	1.63
DOCKING (2)	.9992	1.86
LANDING (3)	.9956	10.22
TOTAL RELIABILITY	.9570	100.00

SYSTEM OR FUNCTION	FERRY/SUPPLY RELIABILITY (LAUNCH–FERRY LANDING)	PERCENT OF TOTAL MISSION RELIABILITY	FERRY/SUPPLY RELIABILITY (LAUNCH – MOORING)	PERCENT OF TOTAL UNRELIABILITY
GUIDANCE & CONTROL	.9667	74.83	.9688(1)	84.49
ENVIRONMENTAL CONTROL	.9978	4.94	.9978	5.99
ELECTRICAL POWER	.9974	5.84	.9982	4.90
COMMUNICATION (VOICE & GROUND TRACKING)	.9998	.45	.9998	.54
INSTRUMENTATION	.9998	.45	.9999	.27
SEQUENTIALS & RETRO-ROCKETS	.9994	1.35	.9996	1.09
DOCKING (2)	.9990	2.25	.9990	2.72
LANDING (3)	.9956	9.89	N/A	
TOTAL RELIABILITY	.9565	100.00	.9633	100.00

SYSTEM OR FUNCTION	STRIPPED GEMINI SUPPLY RELIABILITY (LAUNCH – MOORING)	PERCENT OF TOTAL UNRELIABILITY	SPECIFIC DESIGN SUPPLY RELIABILITY (LAUNCH – MOORING)	PERCENT OF TOTAL UNRELIABILITY
GUIDANCE & CONTROL	.9614	81.32	.9622	80.83
ENVIRONMENTAL CONTROL	.9999	.21	.9998	.43
ELECTRICAL POWER	.9993	1.49	.9993	1.51
COMMUNICATION (VOICE & GROUND TRACKING)	.9934	14.01	.9934	14.22
INSTRUMENTATION	.9998	.42	.9998	.43
SEQUENTIALS & RETRO-ROCKETS	.9996	.85	.9996	.86
DOCKING(2)	.9992	1.70	.9992	1.72
LANDING(3)	N/A		N/A	
TOTAL RELIABILITY	.9529	100.00	.9536	100.00

NOTES:
1. THE FERRY/SUPPLY PROPULSION ESTIMATES ASSUME A SATURN IB LAUNCH. A SATURN I LAUNCH WOULD ALTER THE TOTAL MISSION RELIABILITY IN THE THIRD DECIMAL PLACE.
2. A NOSE DOCK – NOSE MOOR MODE WAS ASSUMED FOR THE FERRY AND UNMANNED SUPPLY SPACECRAFT. A NOSE DOCK – SIDE MOOR OR AFT DOCK – AFT MOOR ASSUMPTION WOULD ALTER THE TOTAL MISSION ESTIMATE IN THE THIRD DECIMAL PLACE. AN AFT DOCK – AFT MOOR MODE WAS ASSUMED FOR THE FERRY/SUPPLY.
3. A PARACHUTE LANDING WAS ASSUMED.

11. WEIGHT ANALYSES

Weight analyses for the spacecraft, and for a number of variations are presented in this section.

The spacecraft, other than the specifically designed Unmanned Supply Spacecraft are modifications of the two-day-rendezvous Gemini. The Specific Design does, however, make maximum use of Gemini equipments.

Derivations for the various spacecraft systems are presented in Section 11.1. The 31 August 1963 Gemini Project Weight Status (Reference 11-1) is used as the base point for all derivations.

Structure for the Ferry/Supply versions is sized from the cargo capability for Saturn IB. Systems are compatible with either Saturn I or IB launch, except that different onboard propulsion systems are used.

The weight aspects of alternate systems are presented in Section 11.2. These alternates can be divided into three categories:

A. Alternates that increase spacecraft weight.

B. Alternates that reduce spacecraft weight.

C. Miscellaneous alternates that increase or reduce launch vehicle capability.

"Adequate" weight margins are derived in Section 11.3, based on the changes to and the design status of Gemini. The past growth history of McDonnell vehicles is used to derive the adequate weight margins. The adequate margins are compared with available weight margins for each of the spacecraft.

11.1 Weight Derivations

11.1.1 Structural System - A structural system weight comparison is shown in Table 11.1-1. The "basepoint" summary shown in the table and utilized throughout this report is based on the 31 August 1963 Gemini Project Weight Status Report (Reference 11-1).

A structural weight derivation for the Ferry versions is shown in Table 11.1-2. Only minor modifications to Gemini structure are required, except for the aft

TABLE 11.1-1
STRUCTURAL SYSTEM WEIGHT COMPARISON

	WEIGHT – (LB)								
	BASE POINT GEMINI	F-1	F-2	F-3	FS-1	FS-2	FS-3	US-1	US-2
RE-ENTRY MODULE	(1654)	(1700)	(1729)	(1715)	(1729)	(1758)	(1744)	(808)	(882)
BASIC STRUCTURE	(882)	(897)	(926)	(896)	(926)	(955)	(925)	(620)	(782)
SKIN AND STIFFENERS	45	50	45	45	50	45	45	41	167
STRINGERS	48	49	48	48	69	68	68	34	63
AFT PRESSURE BULKHEAD	85	85	85	85	85	85	85	85	52
FORWARD PRESSURE BULKHEAD	16	16	16	16	16	16	16	24	
RINGS	87	92	87	86	101	96	95	60	224
TENSION TIES	11	11	11	11	11	11	11	11	
SEPARATION RINGS	4	4	4	4	4	4	4		
CABIN WALLS	54	54	54	54	54	54	54	54	
HATCHES	279	279	309	294	279	309	294	80	
WINDOWS AND DOORS	86	86	86	86	86	86	86	86	
MISCELLANEOUS STRUCTURE	145	149	159	145	149	159	145	145	276
FLOTATION PROVISIONS	22	22	22	22	22	22	22		
SECONDARY STRUCTURE	(448)	(461)	(461)	(450)	(461)	(461)	(450)	(188)	(100)
SHINGLES	240	253	253	253				78	39
INSULATION	169	169	169	159				71	41
MISCELLANEOUS	7	7	7	6				7	
HOIST FITTINGS	8	8	8	8				8	
PAINT	6	6	6	6				6	
INSTALLATION & MISC	18	18	18	18				18	20
HEAT SHIELD	(324)	(342)	(342)	(369)	(342)	(342)	(369)		
ABLATION MATERIAL	239	257	257	257					
BACKUP	77	77	77	77					
MOUNTING PROVISIONS	8	8	8	35					
ADAPTER	(410)	(422)	(422)	(670)	(1785)	(1785)	(2053)	(505)	(*)
BASIC STRUCTURE	(217)	(217)	(217)	(288)				(217)	
SKIN AND STIFFENERS	113	113	113	148					
RINGS	68	68	68	104					
SPLICE & SEPARATE PROVISIONS	36	36	36	36					
SECONDARY STRUCTURE	(193)	(205)	(205)	(382)				(288)	*ADAPTER INTEGRAL WITH CAPSULE
TENSION TIES	17	17	17	17					
SPEARATION PROVISIONS	41	41	41	41					
EQUIPMENT ACCESS DOORS	25	25	25	25					
MISCELLANEOUS STRUCTURE	69	81	81	258	REF. TABLE 11.1-3	REF. TABLE 11.1-3	REF. TABLE 11.1-3		
FAIRINGS	15	15	15	15					
PAINT	24	24	24	24					
RIVETS, NUTS, BOLTS, MISC.	2	2	2	2					
TOTAL STRUCTURE	(2064)	(2122)	(2151)	(2385)	(3514)	(3543)	(3797)	(1313)	(882)

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

TABLE 11.1-2
STRUCTURAL WEIGHT DERIVATION
FERRY SPACECRAFT

ITEM	DERIVATION	WEIGHT – LB. F-1	F-2	F-3
RE-ENTRY MODULE		(1700)	(1729)	(1715)
GEMINI RE-ENTRY MODULE STRUCTURE		(1654)	(1654)	(1654)
BASIC STRUCTURAL CHANGES		(+15)	(+44)	(+14)
SKIN	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+5		
STRINGERS	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+1		
RINGS	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+5		
	REMOVE NOSE DOCKING PROVISIONS			-1
HATCHES	ADD AUXILIARY HATCH (ENCLOSED CREW TRANSFER)		+30	
	ADD HATCH IN AFT PRESSURE BULKHEAD			+15
MISCELLANEOUS STRUCTURE	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+4		
	ADD SIDE MOORING FITTINGS		+14	
SECONDARY STRUCTURAL CHANGES		(+13)	(+13)	(+ 2)
INSULATION	REMOVE NOSE DOCKING PROVISIONS			-10
MISCELLANEOUS	REMOVE NOSE DOCKING PROVISIONS			-1
SHINGLES	INCREASE R & R, RCS, SHINGLES	+13	+13	+13
HEAT SHIELD CHANGES		(+18)	(+18)	(+45)
ABLATION MATERIAL	REFERENCE SECTION 8.5. 18 LB. ADDITIONAL REQUIRED FOR RE-ENTRY FROM PERIGEE OF 87-250 NA.MI. ORBIT WITH 5 GEMINI RETRO-ROCKETS.	+18	+18	+18
MOUNTING PROVISIONS	ADD HATCH IN HEAT SHIELD			+27
ADAPTER		(422)	(422)	(670)
GEMINI ADAPTER STRUCTURE		(410)	(410)	(410)
BASIC STRUCTURAL CHANGES		(0)	(0)	(+71)
SKIN AND STIFFENERS	ADAPTER EXTENSION FOR DOCKING RING			+35
RINGS	ADAPTER EXTENSION FOR DOCKING RING			+36
SECONDARY STRUCTURAL CHANGES		(+12)	(+12)	(+189)
MISCELLANEOUS STRUCTURE	BLAST SHIELD INCREASE DUE TO ADDITIONAL RETRO ROCKETS	+12	+12	+12
	TUNNEL ADDED			+77
	HATCH (TUNNEL TO STATION)			+15
	TUNNEL WINDOW			+30
	MISCELLANEOUS (TUNNEL)			+15
	DOCKING RING ADDED			+40
TOTAL STRUCTURE (FERRY VERSIONS)		(2122)	(2151)	(2385)

F-1 = NOSE DOCK-NOSE MOOR
F-2 = NOSE DOCK-SIDE MOOR
F-3 = AFT DOCK-AFT MOOR

11.1.1 (Continued)

docked - aft moor version for which are added hatches in the heat shield and large pressure bulkhead, and a pressurized tunnel to the adapter section.

A structural weights derivation for the Ferry/Supply Spacecraft is shown in Table 11.1-3. The data used in the table are based on extrapolations of Gemini values. Weights obtained from a detailed preliminary structural analysis of the adapter are quite close to the extrapolated values. Structural weights apply both to Saturn I and Saturn IB launches. Structural weight could be reduced by approximately 300 pounds if the adapter were designed specifically for a Saturn I launch, however.

A structural weight derivation for the Striped Gemini Unmanned Supply Spacecraft is shown in Table 11.1-4. This derivation consists primarily of deleting items of Gemini structure not needed for the unmanned launch. For example, the Gemini hatches and actuation mechanisms are deleted and replaced with simplified, bolted down panels. Since the Gemini hatches are designed by loads, due to opening at maximum q for seat ejection, a significant weight reduction can be accomplished by using fixed panels.

A structural derivation for the Unmanned Supply Spacecraft Specific Design is contained in Table 11.1-5. The derivation is based primarily on preliminary structural analysis and comparison with empirical estimation criteria.

11.1.2 Stabilization and Control Electronics - A weight comparison for the stabilization and control electronic equipment is shown in Table 11.1-6. The Ferry and Ferry/Supply equipment is the same as Gemini. The Unmanned Supply system, however, does not need re-entry control, since the spacecraft is allowed to burn during re-entry. In addition, the digital computer is deleted and a separate digital electronics unit installed to allow rendezvous and docking control from the space station. The changes are reflected in Table 11.1-7.

TABLE 11.1-3
STRUCTURAL WEIGHT DERIVATION
FERRY/SUPPLY SPACECRAFT

ITEM	DERIVATION	WEIGHT – LB. FS-1	FS-2	FS-3
RE-ENTRY MODULE		(1729)	(1758)	(1744)
GEMINI RE-ENTRY MODULE STRUCTURE		(1654)	(1654)	(1654)
BASIC STRUCTURAL CHANGES		(+44)	(+73)	(+43)
SKIN	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+ 5		
STRINGERS	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+ 1		
	NOSE BEEF-UP FOR ESCAPE TOWER	+20	+20	+20
RINGS	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+5		
	NOSE BEEF-UP FOR ESCAPE TOWER	+9	+9	+9
	REMOVE NOSE DOCKING PROVISIONS			-1
HATCHES	ADD HATCH IN HATCH (ENCAPSULATED TRANSFER)		+30	
	ADD HATCH IN AFT PRESSURE BULKHEAD			+15
MISCELLANEOUS STRUCTURE	NOSE BEEF-UP FOR ARTIFICIAL GRAVITY	+4		
	ADD SIDE MOORING FITTINGS		+14	
SECONDARY STRUCTURAL CHANGES		(+13)	(+13)	(+2)
INSULATION	REMOVE NOSE DOCKING PROVISIONS			-10
MISCELLANEOUS	REMOVE NOSE DOCKING PROVISIONS			-1
SHINGLES	INCREASE R & R, RCS, SHINGLES	+13	+13	+13
HEAT SHIELD CHANGES		(+18)	(+18)	(+45)
ABLATION MATERIAL	REFERENCE SECTION 8.5.2. 18 LB. ADDITIONAL REQUIRED FOR RE-ENTRY FROM PERIGEE OF 87-250 NA.MI. ORBIT WITH 5 GEMINI RETRO-ROCKETS.	+18	+18	+18
MOUNTING PROVISIONS	ADD HATCH IN HEAT SHIELD			+27
ADAPTER		(1785)	(1785)	(2053)
GEMINI ADAPTER STRUCTURE		(410)	(410)	(410)
ADAPTER EXTENSION		(1363)	(1363)	(1631)
BASIC SHELL	1.03 LB./FT.^{2} x 308 FT.^{2} x 1.1 NON-OPTIMUM	348	348	348
HIGH EMISSIVITY COATING	.05 LB./FT.^{2} x 308 FT.^{2}	15	15	15
PAINT	.17 LB./FT.^{2} x 308 FT.^{2}	52	52	52
RING UPPER	S/A GEMINI	22	22	22
RING LOWER	22 LB. (GEMINI) x 154/120 (DIA. RATIO)	28	28	28
SHAPED CHARGE	154 IN. DIA. @ .053 LB./IN.	26	26	26
PRESSURIZED CARGO COMPT.				
SIDEWALL	1.25 LB./FT.^{2} x 91.5 FT.^{2}	104	104	104
BULKHEADS	1.25 LB./FT.^{2} x 39.2 FT.^{2}	49	49	49
INSULATION	.25 LB./FT.^{2} x 130.7 FT.^{2}	34	34	34
BULKHEAD RINGS	2-60 IN. DIA. @ .1 LB./IN.	38	38	38
MISCELLANEOUS		5	5	5
CARGO SUPPORT STRUCTURE	$.875(\text{CARGO})^{.65} = .875(20{,}000)^{.65}$	550	550	550
DOCKING RING		92	92	92
TUNNEL ASSEMBLY				
TUNNEL				137
HATCH				15
ADAPTER EXTENSION				71
WINDOW				30
MISCELLANEOUS				15
INCREASED BLAST SHIELD ABLATION MATERIAL		(12)	(12)	(12)
TOTAL STRUCTURE		(3514)	(3543)	(3797)

FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR

TABLE 11.1-4

STRUCTURAL WEIGHT DERIVATION
UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI

ITEM	DERIVATION	WEIGHT LB.
RE-ENTRY MODULE		(808)
GEMINI STRUCTURE		(1654)
BASIC STRUCTURAL CHANGES		(–262)
SKIN	REMOVE RCS SECTION	–4
STRINGERS	REMOVE RCS SECTION	–14
FORWARD PRESSURE BULKHEAD	BEEF-UP FOR HATCH	+8
RINGS	REMOVE RCS SECTION	–27
SEPARATION RINGS	REMOVE SEPARATION PROVISIONS AT STATION 192	–4
HATCHES		
	REMOVE HATCH STRUCTURE	–172
	ADD BOLTED IN PLACE PANELS @ 1.2 LB./FT.^{2}	+30
	REMOVE HATCH ACTUATION MECHANISM	–57
FLOTATION PROVISIONS	NOT REQUIRED	–22
SECONDARY STRUCTURAL CHANGES		(–260)
SHINGLES		
	REMOVE RCS SECTION	–41
	CHANGE FROM RENE TO TITANIUM ON CONICAL SECTION	–61
	CHANGE FROM BERYLLIUM TO TITANIUM ON NOSE SECTION	–60
INSULATION		
	REMOVE RCS SECTION	–22
	REDUCE CONICAL SECTION AND R & R	–76
HEAT SHIELD CHANGES	REMOVE HEAT SHIELD	(–324)
ADAPTER		(505)
GEMINI ADAPTER STRUCTURE		(410)
BASIC STRUCTURAL CHANGES	NONE	(0)
SECONDARY STRUCTURAL CHANGES		(+95)
TENSION TIES	REMOVE PYROTECHNIC SYSTEM	–10
SEPARATION PROVISIONS	DELETE RETRO SECTION SEPARATION	–19
MISCELLANEOUS STRUCTURE	REMOVE MISCELLANEOUS HARDWARE	–26
	ADD CARGO SUPPORT STRUCTURE	+150
TOTAL STRUCTURE		(1313)

TABLE 11.1-5

STRUCTURAL WEIGHT DERIVATION
UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

ITEM	DERIVATION	WEIGHT LB.
BASIC STRUCTURE		(782)
SKIN AND STIFFENERS	.020 TITANIUM SKIN x 1.2 NON-OPTIMUM	167
STRINGERS	8 @ .020 Ti x 156 IN. LONG, 16 @ .020 Ti x 156IN. LONG	
	24 @ .020 Ti x 80 IN. LONG, 1.2 NON-OPTIMUM	63
BULKHEAD	41 IN. RADIUS x .010 TITANIUM	52
RINGS		
	CONICAL SECTION (6), .030 Ti x 3.0 IN. WIDTH	20
	CONICAL SECTION (UNPRESSURIZED)–(3), .082 Ti x 7.1 IN. WIDTH	93
	CYLINDRICAL SECTION (1), .082 Ti x 7.1 IN. WIDE	37
	CONICAL TO ADAPTER (1), .082 Ti x 7.1 IN. WIDE	37
	ADAPTER TO BOOSTER (1), .082 Ti x 7.1 IN. WIDE	37
MISCELLANEOUS STRUCTURE		
	DOCKING CONE (INCLUDES BASIC STRUCTURE, INSULATION, NOSE COVER, HATCH IN FORWARD PRESSURE BULKHEAD, TUNNEL	67
	CARGO SUPPORT STRUCTURE (REFER TO FIGURE 11.1-2)	200
	MISCELLANEOUS	9
SECONDARY STRUCTURE		(100)
SHINGLES	.016 TITANIUM x 1.2 NON-OPTIMUM	39
INSULATION	3.0 LB./FT.^{3} INSULATION x 1 IN. THICK ON SIDES AND BULKHEADS	41
MISCELLANEOUS	SHAPE CHARGE, BOOSTER SEPARATE, SAME AS GEMINI	20
TOTAL STRUCTURE*		(882)

*INCLUDES INTEGRAL ADAPTER

TABLE 11.1-6
GUIDANCE, CONTROL, RE-ENTRY CONTROL SYSTEM WEIGHT COMPARISON

ITEM	WEIGHT – LB.			
	BASE POINT GEMINI	F-1 F-2 F-3	FS-1 FS-2 FS-3	US-1 US-2
RE-ENTRY MODULE	(473)	(473)	(473)	(233)
MANUAL CONTROLS	(11)	(11)	(11)	(0)
AUTOMATIC CONTROLS	(267)	(267)	(267)	(233)
ACME SYSTEM	30			30
INERTIAL GUIDANCE	179			145
HORIZON SENSOR SYSTEM	20			20
MOUNTING AND INSTALLATION	16			16
ELECTRICAL	22			22
RE-ENTRY CONTROL SYSTEM	(195)	(195)	(195)	(0)
PRESSURANT SYSTEM	29			
OXIDIZER SYSTEM	54			
FUEL SYSTEM	45			
THRUSTERS	38			
HEATERS	4			
MOUNTING AND INSTALLATION	11			
ELECTRICAL	14			

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

TABLE 11.1-7
GUIDANCE AND CONTROL SYSTEM WEIGHT DERIVATION UNMANNED SUPPLY SPACECRAFT

ITEM	DERIVATION	WEIGHT – LB.
GEMINI GUIDANCE, CONTROL AND RE-ENTRY CONTROL SYSTEM		(473)
MANUAL CONTROL CHANGES	DELETE – NOT REQUIRED FOR UNMANNED	(-11)
AUTOMATIC CONTROLS		(-34)
INERTIAL GUIDANCE	DELETE COMPUTER-(MOVED TO STATION)	-60
	ADD DIGITAL ELECTRONICS FOR STATION CONTROL	+30
	REMOVE DATA INSERT AND READOUT PANELS	-4
RE-ENTRY CONTROL SYSTEM	DELETE – NO RE-ENTRY	(-195)
TOTAL		(233)

11.1.3 Retrograde System - A retrograde system weight comparison is presented in Table 11.1-8. The major changes from Gemini are:

A. Two additional Gemini retrorockets are added.

B. Two 11-pound solid propellant separation rockets are added.

C. Retrograde rockets and associated items are deleted from the Unmanned Supply Spacecraft. After separating from the space station with the separation rockets, the spacecraft will re-enter following orbit decay.

TABLE 11.1-8
RETROGRADE SYSTEM
WEIGHT COMPARISON

ITEM	WEIGHT – LB.				
	BASE POINT GEMINI	F-1 F-2 F-3	FS-1 FS-2 FS-3	US-1	US-2
RE-ENTRY MODULE	(21)	(21)	(21)	(0)	(25)
CIRCUITRY	21	21	21		
SEPARATION ROCKETS					22
SEPARATION ROCKET INSTALLATION					3
ADAPTER	(341)	(519)	(544)	(45)	(*)
RETROGRADE ROCKET MOTORS	263	395	395		
INSTALLATION	34	51	51	20	
CIRCUITRY	10	14	14		
BLAST PROTECTION	31	31	31		
MISCELLANEOUS	3	3	3		
SEPARATION ROCKETS		22	43	22	
SEPARATION ROCKET INSTALLATION		3	7	3	
*ADAPTER INTEGRAL WITH SPACECRAFT					
TOTAL RETROGRADE SYSTEM WEIGHT	(362)	(540)	(565)	(45)	(25)

11.1.4 Landing System - The landing system for the Ferry and Ferry/Supply Spacecraft is identical to that in Gemini. A landing system is not required for Unmanned Supply versions.

11.1.5 Instrumentation and Navigation System - A weight comparison of the Instrumentation and Navigation systems is shown in Table 11.1-9. The Ferry and Ferry/Supply versions use the same system as Gemini. In the Unmanned Supply

11.1.5 (Continued)

Spacecraft all displays are removed except the time reference system.

TABLE 11.1-9

INSTRUMENTATION AND NAVIGATION SYSTEM
WEIGHT COMPARISON

ITEM	WEIGHT – LB.			
	BASE POINT GEMINI	F–1 F–2 F–3	FS–1 FS–2 FS–3	US–1 US–2
FLIGHT AND MISSION INDICATOR	26	26	26	–
WARNING AND SEQUENCE LIGHTS	5	5	5	–
ENVIRONMENTAL INDICATORS	5	5	5	–
ELECTRICAL INDICATORS	4	4	4	–
PROPULSION INDICATORS	6	6	6	–
RENDEZVOUS SYSTEM INDICATORS	6	6	6	–
TIMING SYSTEM	11	11	11	11
PANELS, CONSOLES	34	34	34	–
PLUMBING & CIRCUITRY	38	38	38	–
MAPS – CHART BOARD	2	2	2	–
TOTAL INSTRUMENTATION AND NAVIGATION	(137)	(137)	(137)	(11)

FS–1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS–2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS–3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
F–1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F–2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F–3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
US–1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US–2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

11.1.6 Electrical Power System - Electrical Power System weights are compared in Table 11.1-10.

The Ferry and Ferry/Supply Spacecraft utilize an identical electrical power system. Changes in this system from Gemini are shown in Table 11.1-11. Electrical power systems for the Unmanned Supply Spacecraft are derived in Table 11.1-12.

The Ferry and Ferry/Supply Spacecraft utilize Gemini fuel cells with 18 pounds of reactants. The Unmanned Supply Spacecraft, however, due to lower power requirements, utilize multiples of the Gemini main batteries.

For the post separation through re-entry phase of the ferry mission, automatically activated silver-zinc batteries are used. Specific power for batteries of this type, based on total system weight, is 30-35 watt-hours per pound, compared

TABLE 11.1-10

ELECTRICAL POWER SYSTEM WEIGHT COMPARISON

ITEM	BASE POINT GEMINI	WEIGHT (LB.) F-1 F-2 F-3	FS-1 FS-2 FS-3	US-1	US-2
RE-ENTRY MODULE	(282)	(333)	(333)	(299)	(240)
BATTERIES	105	150		205	165
FUEL CELL CONTROLS	9	9			
GROUND UMBILICAL	3	3		3	6
SWITCHES & MISC.	33	33		26	21
WIRING	85	85		30	15
INTERIOR LIGHTS	3	3			
NOSE FAIRING SEPARATION	2	2		2	2
R & R MODULE SEPARATION	11	11			
ADAPTER SEPARATION	12	12			
BOOSTER SEPARATION	3	3		3	7
MOUNTING AND INSTALLATION	16	22		30	24
ADAPTER	(393)	(459)	(459)	(17)	(—)
FUEL CELLS	152	152			
FUEL CELLS REACTANTS	52	18			
CONTAINERS	61	61			
PLUMBING	10	10			
GROUND UMBILICAL	3	3		3	
FUEL CELL WIRING	40	40			
ADAPTER SEPARATION	9	9			
BOOSTER SEPARATION	4	4		4	
MOUNTING AND INSTALLATION	62	75		10	
BATTERIES		87			
TOTAL	(675)	(792)	(792)	(316)	(240)

FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR

TABLE 11.1-11

ELECTRICAL POWER SYSTEM WEIGHT DERIVATION
FERRY AND FERRY/SUPPLY SPACECRAFT

ITEM	DERIVATION	WEIGHT LB.
RE-ENTRY MODULE		(333)
GEMINI		282
BATTERY WEIGHT CHANGES		
MAIN BUS	REMOVE GEMINI BATTERIES, 4 @ 20 LB. EA. (MANUALLY ACTIVATED)	−80
	ADD 2 @ 30 LB. EA., 2 @ 34 LB. EA., (AUTOMATICALLY ACTIVATED)	+128
SQUIB BUS	REMOVE GEMINI BATTERIES, 3 @ 8.4 LB. EA.	−25
	ADD 3 @ 7.5 LB. EA. (AUTOMATICALLY ACTIVATED)	+22
MOUNTING AND INSTALLATION WEIGHT CHANGES	ADD MOUNTING FOR BATTERIES	+6
ADAPTER		(459)
GEMINI		393
FUEL CELL REACTANT WEIGHT CHANGE	52 LB. IN 2 DAY GEMINI, 18 LB. RQD. FOR FERRY AND FERRY SUPPLY	−34
BATTERY WEIGHT CHANGE		
MAIN BUS	ADD 3 @ 22 LB. EA. AUTOMATICALLY ACTIVATED, FOR POST SEPARATION ORBITING POWER.	+66
	ADD 1 @ 4 LB. MANUALLY ACTIVATED, FOR FUEL CELL BACKUP, LAUNCH PHASE ABORT	+4
SQUIB BUS	ADD 3 @ 5.5 LB. EA. (MANUALLY ACTIVATED)	+17
MOUNTING AND INSTALLATION WEIGHT CHANGES	ADD MOUNTING FOR BATTERIES	+13
TOTAL ELECTRICAL POWER SYSTEM (FERRY AND FERRY/SUPPLY)		(792)

TABLE 11.1-12

ELECTRICAL POWER SYSTEM WEIGHT DERIVATION
UNMANNED SUPPLY SPACECRAFT

ITEM	DERIVATION	WEIGHT – LB. US–1	WEIGHT – LB. US–2
RE-ENTRY MODULE		(299)	(240)
GEMINI		282	282
BATTERY WEIGHT CHANGES			
MAIN BUS	GEMINI HAS 4 @ 20 LB. EA. MANUALLY ACTIVATED		
	ADD 5 GEMINI TYPE FOR US–1	+100	
	ADD 3 GEMINI TYPE FOR US–2		+60
SQUIB BUS	NO CHANGE TO GEMINI		
FUEL CELL CONTROL	NOT REQUIRED	–9	–9
SWITCHES AND MISC.			
	REMOVE GEMINI ITEMS	–32	–32
	ADD EST. REQD. FOR US–1 AND US–2	+15	+10
WIRING	REMOVE GEMINI WIRE	–85	–85
	ADD EST. REQD. FOR US–1 AND US–2	+30	+15
	ADD 150 V.A. INVERTER FOR GIMBAL SYSTEM	+10	+10
MISC. ITEMS	INTERIOR LIGHTS, R & R AND ADAPTER SEPARATION NOT REQUIRED	–26	–26
MOUNTING AND INSTALLATION	ADD MOUNTING FOR ADDITIONAL BATTERIES	+14	+8
LAUNCH VEHICLE SEPARATION	US–2 ADAPTER INTEGRAL WITH RE-ENTRY MODULE		+4
GROUND UMBILICAL	US–2 ADAPTER INTEGRAL WITH RE-ENTRY MODULE		+3
ADAPTER		(17)	(*)
THE FOLLOWING GEMINI ITEMS ARE REQUIRED FOR US–1:			
GROUND UMBILICAL	SAME AS GEMINI	3	
LAUNCH VEHICLE SEPARATION	SAME AS GEMINI	4	
MOUNTING AND INSTALLATION	10 LB. OF GEMINI DIFFICULT TO REMOVE	10	
TOTAL ELECTRICAL POWER SYSTEM		(316)	(240)

*US–2 HAS INTEGRAL ADAPTER

US–1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US–2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

11.1.6 (Continued)

to 54 watt-hours per pound for Gemini. This factor, plus the fact that fuel cells are not utilized for post separation power, is the primary cause for the weight increases reflected in Table 11.1-11.

The power system for the Unmanned Supply Spacecraft, Table 11.1-12, includes a 150 V.A. inverter for the gimballed self-inject engine. The Stripped Gemini utilizes a modified Gemini Environmental Control System (ECS) which requires more power than the simplified ECS provided in the Specific Design.

11.1.7 Communications System - Communications system weights are compared in Table 11.1-13.

TABLE 11.1-13

COMMUNICATIONS SYSTEM WEIGHT COMPARISON

ITEM*	WEIGHT – LB.					
	BASE POINT GEMINI	F-1 F-2 F-3	FS-1 FS-2	FS-3	US-1	US-2
RE-ENTRY MODULE	(56)	(56)	(56)	(56)	(49)	(55)
U.H.F. VOICE	7					
C-BAND BEACON	11				11	11
CONTROL CENTER	7					
MISC. COMPONENTS	8				8	9
MOUNTING	9				9	9
CIRCUITRY	14				14	19
ACQUISITION LIGHTS					7	7
ADAPTER	(28)	(28)	(32)	(28)	(6)	(*)
H.F. VOICE	4		4			
S-BAND BEACON	9		9			
ACQUISITION AID	1		1		1	
MISC. EQUIPMENT	2		6			
CIRCUITRY	12		12		5	
TOTAL COMMUNICATIONS SYSTEM WEIGHT	(84)	(84)	(88)	(84)	(55)	(55)

*INTEGRAL ADAPTER FOR US-2; WEIGHT INCLUDED UNDER MISC. COMPONENTS

FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN
F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR

11.1.7 (Continued)

The systems for the Ferry Spacecraft and the aft dock - aft moor Ferry/Supply Spacecraft are the same as for Gemini. The system for the other two Ferry/Supply versions is similar except that a docking command receiver is added to permit remotely controlled aft docking from the station.

The Unmanned Supply Spacecraft systems are similar to Gemini except that the S-Band beacon and voice communications are deleted and two lights and a docking receiver are added to facilitate acquisition by the station.

11.1.8 Environmental Control and Equipment Cooling Systems - ECS and Equipment Cooling System weights are compared in Table 11.1-14.

Weight derivations for the Ferry and Ferry/Supply versions are presented in Table 11.1-15. The volume and pressure of the gaseous O_2 bottle in the re-entry module is increased to provide an additional O_2 capability of 8 pounds. These bottles, utilized on Gemini for the re-entry phase only, are used for three hours of post-separation orbiting and for the longer re-entry on Ferry and Ferry/Supply missions. The cryogenic oxygen stored in the adapter is used only during the launch-to-docking phase and is accordingly reduced compared to Gemini.

An egress system is provided for each of the Ferry and Ferry/Supply versions. Two 1.25-inch diameter ECS extension lines, 20 feet long, are required for transfer through space for nose-moored versions. The ECS extension lines used for crew transfer with the side-mooring versions are assumed included in the tunnel attached to the space station. For the longer tunnel in the aft-docked Ferry/Supply versions, two 1.25-inch diameter lines, 45 feet long, are required. Backup to the ECS extension lines is provided by pressurizing the tunnel for crew transfer or by portable oxygen supplies for extravehicular transfer.

An estimated 44 pounds is included on Ferry and Ferry/Supply Spacecraft for heaters and special coatings to control equipment temperature during the standby phase.

TABLE 11.1-14
ENVIRONMENTAL CONTROL SYSTEM WEIGHT COMPARISON

ITEM	WEIGHT – LB.						
	BASE POINT GEMINI	F-1, FS-1	F-2, FS-2	F-3	FS-3	US-1	US-2
RE-ENTRY MODULE	(298)	(374)	(338)	(353)	(372)	(53)	(172)
PRESSURE BREATHING							
VALVING	9	9	9	9	9		
COMPRESSORS	9	9	9	9	9		
CONTROLS	10	10	10	10	10		
MOUNTINGS	4	4	4	4	4		
RENOVATING SYSTEM							
SOLIDS TRAP	1	1	1	1	1		
CO_2 ABSORBER	50	50	50	50	50		
VALVING	13	13	13	13	13		
DUCTS AND CLAMPS	16	16	16	16	16		
MOUNTS	6	6	6	6	6		
COOLING SYSTEM							
HEAT EXCHANGERS	29	29	29	29	29		
COOLING FLUIDS	16	16	16	16	16	10	50
COLD PLATES	32	32	32	32	32	32	82
LINES & VALVES	10	10	10	10	10	10	11
MISCELLANEOUS	1	1	1	1	1	1	29
POWER SUPPLIES	11	11	11	11	11		
SECONDARY O_2							
OXYGEN	14	22	22	22	22		
OXYGEN PACKAGES	41	73	73	73	73		
LINES, MOUNTS	4	4	4	4	4		
CIRCUITRY	11	11	11	11	11		
MISCELLANEOUS	11	11	11	11	11		
EGRESS SYSTEM		36		15	34		
ADAPTER	(254)	(290)	(290)	(292)	(293)	(129)	*
PRESSURE BREATHING							
OXYGEN	16	8	8	10	11		
OXYGEN BOTTLE	13	13	13	13	13		
OXYGEN VALVES	9	9	9	9	9		
DUCTS, CLAMPS, ETC.	5	5	5	5	5		
COOLING SYSTEM							
HEAT EXCHANGERS	24	24	24	24	24	2	
COOLING FLUID	40	40	40	40	40	23	
PUMPS	60	60	60	60	60	42	
COLD PLATES	9	9	9	9	9	9	
LINES AND CLAMPS	22	22	22	22	22	20	
VALVES & MISCELLANEOUS	10	54	54	54	54	5	
BLAST SHIELD	18	18	18	18	18		
CIRCUITRY	16	16	16	16	16	16	
MODULAR STRUCTURE	10	10	10	10	10	10	
MOUNTS	2	2	2	2	2	2	
TOTAL E.C.S. WEIGHT	(552)	(664)	(628)	(645)	(665)	(182)	(172)

*US-2 ADAPTER IS INTEGRAL WITH RE-ENTRY MODULE

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

TABLE 11.1-15

ENVIRONMENTAL CONTROL SYSTEM WEIGHT DERIVATION
FERRY AND FERRY/SUPPLY SPACECRAFT

ITEM	DERIVATION	WEIGHT – LB. F-1 FS-1	F-2 FS-2	F-3	FS-3
RE-ENTRY MODULE		(374)	(338)	(353)	(372)
GEMINI		298	298	298	298
SECONDARY O_2 WEIGHT CHANGE					
OXYGEN	ADD ADDITIONAL GASEOUS OXYGEN FOR LEAKAGE, 3 HR. POST SEPARATION ORBITING AND LONGER RE-ENTRY.	+8	+8	+8	+8
OXYGEN PACKAGES	VOLUME OF EACH OF TWO TANKS INCREASED FROM 438 $IN.^3$ TO 530 $IN.^3$ AND PRESSURE FROM 5000 PSI TO 7500 PSI.	+32	+32	+32	+32
EGRESS SYSTEM					
ADD PORTABLE O_2 SUPPLY	TO BACK UP HOSES FOR EGRESS TO SPACE	+21			
ADD TRANSFER LINES	2 LINES, 1.25 IN. DIA. X 20 FT. @.375 LB./FT. = 15 LB.	+15		+15	
	2 LINES, 1.25 IN. DIA. x 45 FT. @ .375 LB./FT. = 34 LB.				34
ADAPTER		(290)	(290)	(292)	(293)
GEMINI		254	254	254	254
PRESSURE, BREATHING	OFFLOAD O_2	–8	–8	–6	–5
COOLING SYSTEM WEIGHT CHANGE					
VALVES & MISC.	ADD HEATERS AND COATINGS FOR STANDBY TEMPERATURE CONTROL.	+44	+44	+44	+44
TOTAL ECS WEIGHT		(664)	(628)	(645)	(665)

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR.

11.1.8 (Continued)

ECS requirements for the Unmanned Supply Spacecraft consist primarily of equipment temperature control during the launch-to-docking phase. The system for Stripped Gemini is based on minimum change to Gemini and utilizes the bottom radiator area only (adapter equipment section). Pumps, pump power and cooling fluids are reduced accordingly. In addition functions associated with the manned launch, i.e., cabin fan, suit heat exchanger and launch cooling water are eliminated.

A significantly simplified system is provided for the Specific Design. Cold plates contain refrasil saturated with water. Radiators and pumps are not used. Water is evaporated in the cold plates at a temperature controlled by a pressure relief valve.

11.1.9 Tele-Instrumentation System - A tele-instrumentation system weight comparison is shown in Table 11.1-16. The Gemini system, with R and D measurement functions deleted, is used. Only integral items are deleted, i.e., black boxes or components are not redesigned for reduced capability. The system capability is reduced to approximately 50 percent of Gemini. The remaining capability is still more than twice that required for operational measurements.

11.1.10 Recovery System - The Recovery System for the Ferry and Ferry/Supply Spacecraft is identical to that in Gemini. A recovery system is not required for the Unmanned Supply versions.

11.1.11 Rendezvous System - Rendezvous System weights are compared in Table 11.1-17.

The Ferry and Ferry/Supply versions utilize the Gemini rendezvous system, except that an Agena-type radar transponder system, similar to that used for the Gemini-Agena rendezvous, has been added, to provide capability for backup control from MORL. In addition, nose docking provisions are deleted from the aft docking versions, and aft docking controls added. For the Unmanned Supply Spacecraft the

TABLE 11.1–16

TELE-INSTRUMENTATION SYSTEM WEIGHT COMPARISON

ITEM	(WEIGHT – LB.)				
	BASE POINT GEMINI	F–1 F–2 F–3	FS–1 FS–2 FS–3	US–1	US–2
RE-ENTRY MODULE	(151)	(123)	(123)	(123)	(184)
DATA TRANSMISSION SYSTEM	(44)	(44)	(44)	(44)	(44)
TAPE RECORDER	12	12			
PCM PROGRAMMER	20	20			
MULTIPLEXER HI	3	3			
MULTIPLEXER LO	2	2			
TRANSMITTER-TELEM. MIDDLE	3	3			
TRANSMITTER-TELEM. LO	3	3			
CIRCUITRY	1	1			
POWER CONVERSION AND REGULATION	(14)	(14)	(14)	(14)	(14)
D.C. CONVERTER AND REGULATOR	7	7			
D.C. CONVERTER AND REGULATOR	7	7			
DATA CONDITIONING COMPONENTS	(12)	(12)	(12)	(12)	(12)
DATA PICKUPS	(11)	(11)	(11)	(11)	(14)
COMPONENTS	7	7			10
MOUNTING	1	1			1
CIRCUITRY	3	3			3
BIO-MEDICAL SYSTEM	(20)				
TAPE RECORDER	4				
DC–DC CONVERTER	3				
MEDICAL DATA KIT	10				
MOUNTING	1				
CIRCUITRY	2				
MISC. MOUNTING PROVISIONS	(2)	(2)	(2)	(2)	(21)
CIRCUITRY, MISCELLANEOUS	(48)	(40)	(40)	(40)	(79)
ADAPTER	(100)	(61)	(61)	(61)	(*)
DATA TRANSMISSION SYSTEM	(8)				
MULTIPLEXER – HI	3				
MULTIPLEXER – LO	2				
TRANSMITTER-TELEM.	3				
COMMAND SYSTEM	(24)	(24)	(24)	(24)	
DATA CONDITIONING	(4)				
DATA PICKUPS	(2)	(2)	(2)	(2)	
MISC. COMPONENTS	(3)	(1)	(1)	(1)	
DIPLEXER	2				
UHF EXTENDIBLE ANTENNA	1	1			
MOUNTING & MISCELLANEOUS	(29)	(19)	(19)	(19)	
CIRCUITRY	(30)	(15)	(15)	(15)	
TOTAL TELE-INSTRUMENTATION SYSTEM WEIGHT	(251)	(184)	(184)	(184)	(184)

* US–2 HAS AN INTEGRAL ADAPTER

FS–1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS–2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS–3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US–1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US–2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

11.1.11 (Continued)

TABLE 11.1-17

RENDEZVOUS SYSTEM
WEIGHT COMPARISON

ITEM	WEIGHT – LB.					
	BASE POINT GEMINI	F-1 FS-1	F-2 FS-2	F-3	FS-3	US-1 US-2
RE-ENTRY MODULE	(126)	(166)	(166)	(154)	(154)	(92)
OAMS CIRCUITRY	5	5		5		5
DOCKING PROVISIONS	12	12				12
CONTROLS	3	3		3		3
MOUNTING	1	1		1		1
CIRCUITRY	16	16		16		16
ELECTRONICS						
RENDEZVOUS	68	68		68		
RADAR ENCODER	2	2		2		
INSTALLATION	1	1		1		
CIRCUITRY	18	18		18		15
RADAR TRANSPONDER SYSTEM		40		40		40
ADAPTER	(0)	(0)	(0)	(23)	(28)	(0)
AFT DOCKING CONTROLS	–	–	–	23	28	–
TOTAL RENDEZVOUS SYSTEM WEIGHT	(126)	(166)	(166)	(177)	(182)	(92)

FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR

rendezvous radar is replaced by the Agena radar transponder. Rendezvous is controlled from the ground and from MORL, and docking is manually controlled by an astronaut on MORL.

11.1.12 Crew and Survival Systems - Crew and survival systems weights are shown in Table 11.1-18.

Ferry and Ferry/Supply systems are the same as Gemini, except that food for one day is eliminated due to the reduced catch-up time. In addition, protective coveralls are added for the nose dock - nose moor versions for use during extravehicular crew transfer.

11.1.12 (Continued)

TABLE 11.1-18
CREW AND SURVIVAL SYSTEMS
WEIGHT COMPARISON

ITEM	WEIGHT – LB.					
	BASE POINT GEMINI	F-1	F-2 F-3	FS-1	FS-2 FS-3	US-1 US-2
CREW	360	360	360	360	360	–
PRESSURE SUIT AND GEAR	55	55	55	55	55	–
PERSONAL PARACHUTE	34	34	34	34	34	–
SURVIVAL KIT	54	54	54	54	54	–
EGRESS KIT	43	43	43	43	43	–
HOSE FITTINGS	1	1	1	1	1	–
PERSONAL EFFECTS KIT	19	19	19	19	19	–
RADIATION DETECTION	7	7	7	7	7	7
FOOD	18	14	14	14	14	–
WATER	27	27	27	27	27	–
FLASHLIGHT	1	1	1	1	1	–
RELIEF FACILITIES	1	1	1	1	1	–
EJECTION SEAT INSTALLATION	196	196	196	196	196	–
PYROTECHNICS	3	3	3	3	3	–
CATAPULT	57	57	57	57	57	–
EJECTION SEAT BACKUP STRUCTURE	47	47	47	47	47	–
SOLAR RADIATION GARMENTS	–	7	–	7	–	–
TOTAL CREW & SURVIVAL SYSTEMS WEIGHT	(923)	(926)	(919)	(926)	(919)	(7)

FS–1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS–2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS–3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US–1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US–2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN
F–1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F–2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F–3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR

For the unmanned versions, crew and survival items are deleted, except for a radiation monitoring system.

11.1.13 Propulsion System - Propulsion system weights are shown in Table 11.1-19. These systems are in addition to the RCS, retrograde and launch escape propulsion systems, which are discussed separately.

The nose-dock Ferry Spacecraft utilizes the same OAMS as the two-day-rendezvous Gemini. In the aft-dock version a 100-pound-thrust engine is added and one relocated,

TABLE 11.1-19
PROPULSION SYSTEMS WEIGHT COMPARISON

ITEM	WEIGHT – LB.								
	GEMINI BASE POINT	F-1 F-2	F-3	FS-1 FS-2 (S-I)	FS-3 (S-I)	FS-1 FS-2 (S-IB)	FS-3 (S-IB)	US-1	US-2
TOTAL SYSTEM WEIGHT*	1089	1089	1099	3395	3143	4912	4660	5841	5762
THRUST CHAMBERS	(96)	(96)	(105)	(351)	(255)	(396)	(300)	(227)	(227)
ORBIT ATTITUDE & MANEUVER	96	96	105	324	228	324	228	96	96
SELF INJECTION								113	113
STATION MAINTENANCE				27	27	72	72	18	18
INSTALLATION – THRUSTER	(16)	(16)	(17)	(47)	(31)	(52)	(36)	(26)	(26)
PRESSURIZATION SYSTEM	(57)	(57)	(57)	(343)	(325)	(580)	(562)	(374)	(369)
PRESSURANT	4.6			126.5	125	223.5	222	60.8	82.2
TANKAGE									
SELF INJECTION	38							155	155
ORBIT ATTITUDE & MANEUVER				188	186	328	326	57	102
MAINTENANCE & STABILIZATION								57	
VALVING	14			28	14	28	14	44	30
PROPELLANT*	(715)	(715)	(715)	(2117)	(2076)	(3095)	(3054)	(4527)	(4527)
ORBIT ATTITUDE & MANEUVER	715			2117	2076	3095	3054	884	884
SELF INJECTION								3643	3643
TANKAGE-PROPELLANT	(41)	(41)	(41)	(152)	(144)	(288)	(280)	(207)	(173)
SELF INJECTION								95	95
ORBIT ATTITUDE & MANEUVER	41	41	41	152	144	288	280	56	78
MAINTENANCE & STABILIZATION								56	
PROPELLANT VALVING	(10)	(10)	(10)	(21)	(12)	(21)	(12)	(25)	(16)
TUBING	(18)	(18)	(18)	(55)	(36)	(60)	(41)	(32)	(32)
ELECTRICAL	(35)	(35)	(35)	(103)	(68)	(114)	(79)	(60)	(60)
SUPPORT STRUCTURE	(101)	(101)	(101)	(206)	(196)	(306)	(296)	(363)	(332)

*DOES NOT INCLUDE THE STATION MAINTENANCE AND STABILIZATION PROPELLANT (THIS IS CONSIDERED PART OF CARGO).

STATION-KEEPING PROPELLANT	(1838)	(3859)	(912)	(1234)
STATION STABILIZATION	1320	2640	660	880
STATION MAINTENANCE	476	1086	218	313
TRAPPED	42	133	34	41

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN
S-I = SATURN I LAUNCHED
S-IB = SATURN IB LAUNCHED

11.1.13 (Continued)

due to interference with the adapter tunnel, otherwise the system is the same.

Propulsion system derivations for the Ferry/Supply Spacecraft are shown in Tables 11.1-20 and 11.1-21 for Saturn I and IB launches. Propulsion system requirements are higher for Saturn IB launches due to the higher payload.

TABLE 11.1-20

PROPULSION SYSTEM WEIGHT DERIVATION

FERRY/SUPPLY SPACECRAFT

SATURN I LAUNCHED

ITEM	DERIVATION	WEIGHT – LB. COMPOSITE	OAMS (2)
THRUST CHAMBERS (OAMS)	24– 100 LB. GEMINI THRUSTERS AT 9 LB. EA. PLUS 4 – 25 LB. THRUSTERS AT 3 LB. EA.	228	
ORBIT MAINTENANCE	3 – 100 LB. GEMINI THRUSTERS AT 9 LB. EACH	27	
FLY AROUND	GEMINI OAMS THRUSTERS AND INSTALLATION		96
THRUSTER INSTALLATION	THRUSTERS AT 1.0 LB. EA.	31	16
PRESSURANT (NITROGEN)	REFER TO SECTION 7.2	125	1.5
PRESSURANT TANK	1 REQUIRED, 31.04 IN. I.D. SPHERICAL, GEMINI DESIGN CRITERIA, 1.062 NON-OPTIMUM FACTOR	186	
	GEMINI RCS PRESSURE TANK		2.1
PRESSURANT VALVES	MODIFIED GEMINI COMPONENTS	14	14
PROPELLANT (1)	REFER TO SECTION 7.2	2076	41
OXIDIZER TANKS	2 REQUIRED, 37 IN. I.D. SPHERICAL, GEMINI DESIGN CRITERIA, 1.25 NON-OPTIMUM FACTOR GEMINI LARGE RCS TANK	72	4
FUEL TANKS	2 REQUIRED, SAME AS OXIDIZER TANKS	72	
	GEMINI LARGE RCS TANK		4
OXIDIZER VALVES	MODIFIED GEMINI COMPONENTS	2	2
FUEL VALVES	MODIFIED GEMINI COMPONENTS	2	2
MISCELLANEOUS VALVING	SHUT-OFF VALVES AND TUBING CUTTERS	8	5
LINES & MISCELLANEOUS	NUMBER OF THRUSTERS AND LENGTH OF LINES	36	19
MODULE STRUCTURE	STATICAL EQUATION FOR TUBULAR TRUSS SUPPORT WT. = $(.875)\,(\text{WT. SUPPORTED})^{.65}$	196	10
ELECTRICAL	NUMBER AND SIZE OF THRUSTER	68	35
TOTAL (1)	3395	3143	252

(1) DOES NOT INCLUDE STATION KEEPING PROPELLANT 1838 LB. (2) OAMS ON NOSE-DOCKING FERRY

Ferry/Supply spacecraft utilize a composite propulsion system that performs the following functions:

A. Orbit attitude and maneuvering

B. Station orbit maintenance

C. Station stabilization (propellant supply only)

11.1.13 (Continued)

TABLE 11.1-21

PROPULSION SYSTEM WEIGHT DERIVATION
FERRY/SUPPLY SPACECRAFT
SATURN IB LAUNCHED

ITEM	DERIVATION		WEIGHT – LB.	
			COMPOSITE	OAMS②
THRUST CHAMBER OAMS	24–100 LB. GEMINI THRUSTERS AT 9 LB. EA. PLUS 4 – 25 LB. GEMINI THRUSTERS AT 3 LB. EA.		228	
ORBIT MAINTENANCE	8 – 100 LB. GEMINI THRUSTERS AT 9 LB. EA.		72	
FLY AROUND	GEMINI OAMS THRUSTERS AND INSTALLATION			96
THRUSTER INSTALLATION	THRUSTERS AT 1.0 LB. EA.		36	16
PRESSURANT (NITROGEN)	REFER TO SECTION 7.2		222	2
PRESSURANT TANK	4 REQUIRED 23.60 IN. I.D. SPHERICAL, GEMINI DESIGN CRITERIA, 1.062 NON-OPTIMUM		326	
	GEMINI RCS PRESSURE TANK			2
PRESSURANT VALVES	MODIFIED GEMINI COMPONENTS		14	14
PROPELLANT①	REFER TO SECTION 7.2		3054	41
OXIDIZER TANKS	2 REQUIRED, 37 IN. I.D. HEMISPHERICAL ENDS WITH 19 IN. CYLINDRICAL SECTION, GEMINI DESIGN CRITERIA, 1.188 NON-OPTIMUM FACTOR		140	
	GEMINI LARGE RCS TANK			4
FUEL TANKS	2 REQUIRED SAME AS OXIDIZER TANKS		140	
	GEMINI LARGE RCS TANK			4
OXIDIZER VALVES	MODIFIED GEMINI COMPONENTS		2	2
FUEL VALVES	MODIFIED GEMINI COMPONENTS		2	2
MISCELLANEOUS VALVING	SHUT-OFF VALVES AND TUBING CUTTERS		8	5
LINES AND MISCELLANEOUS	NUMBER OF THRUSTERS AND LENGTH OF LINES		41	19
MODULE STRUCTURE	STATICAL EQUATION FOR TUBULAR TRUSS SUPPORT WT. = (.087) (WT. SUPPORTED)$^{.65}$		296	10
ELECTRONICS	NUMBER AND SIZE OF THRUSTER		79	35
TOTAL①		4912	4660	252

① DOES NOT INCLUDE STATION KEEPING PROPELLANT 3859 LB.
② OAMS USED ONLY ON NOSE DOCKING FERRY VERSIONS

In addition, the nose dock ferry versions have a separate "fly around" OAMS for nose docking of the Ferry after the cargo module has been aft moored.

Weight derivations for the Unmanned Supply propulsion systems are shown in Tables 11.1-22 and 11.1-23. Both versions incorporate a self-inject system which uses a 2200-pound thrust Apollo sub-scale gimballed engine. The Stripped Gemini version has two additional propulsion systems. These systems utilize Gemini, or multiples of Gemini, components. One system is for station orbit maintenance and stabilization. The other is for orbit attitude control and maneuvering. The specific design has, in addition to the self-inject system,

TABLE 11.1-22

PROPULSION SYSTEM WEIGHT DERIVATION
UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI

ITEM	DERIVATION	WEIGHT – LB. SELF INJECTION	OAMS	MAINTENANCE AND STABILIZATION
THRUST CHAMBERS	1 – 2200 LB. GIMBAL THRUSTER	113		
	GEMINI THRUSTER SYSTEM		96	
	2 @ 100 LB. GEMINI THRUSTER			18
INSTALLATION – THRUSTER		8	16	2
PRESSURANT	REFER TO SECTION 7.2	14	6	41
PRESSURANT – TANKS	1 @ 29.21 IN. I.D. SPHERICAL GEMINI DESIGN CRITERIA 1.064 NON-OPTIMUM FACTOR	155		
	3 @ GEMINI 15.21 IN. O.D. SPHERICAL TANKS		57	57
PRESSURANT – VALVES	MODIFIED GEMINI COMPONENTS	16	14	14
PROPELLANT (1)	REFER TO SECTION 7.2	3643	884	
OXIDIZER TANKS	1 @ 39.20 IN. I.D. HEMISPHERICAL END WITH 17.5 IN. CYLINDRICAL SECTION, 1.229 NON-OPTIMUM FACTOR	54		
	3 @ GEMINI 20.12 IN. O.D. SPHERICAL TANKS		28	28
FUEL TANKS	1 @ 39.20 IN. I.D. HEMISPHERICAL END WITH 8.4 IN. CYLINDRICAL SECTION, 1.240 NON-OPTIMUM FACTOR	41		
	3 @ GEMINI 20.12 IN. O.D. SPHERICAL TANKS		28	28
OXIDIZER VALVES	MODIFIED GEMINI COMPONENTS	2	2	2
FUEL VALVES	MODIFIED GEMINI COMPONENTS	2	2	2
MISCELLANEOUS VALVES	SHUT-OFF VALVES AND/OR TUBING CUTTERS	1	6	6
LINES AND MISCELLANEOUS	NUMBER OF THRUSTERS AND LENGTH OF LINES	5	19	8
ELECTRICAL	NUMBER AND SIZE OF THRUSTERS	10	35	15
SUPPORT STRUCTURE	STATISTICAL EQUATION FOR TUBULAR TRUSS WEIGHT. SUPPORT WT. = (.875) (WEIGHT SUPPORTED)$^{.65}$	191	88	84
TOTAL (1)	5841	4255	1281	305

(1) DOES NOT INCLUDE STATION KEEPING PROPELLANT 912 LB.

TABLE 11.1-23

PROPULSION SYSTEM WEIGHT DERIVATION

UNMANNED SUPPLY SPACECRAFT–SPECIFIC DESIGN

ITEM	DERIVATION		WEIGHT – LB. SELF INJECTION	WEIGHT – LB. COMPOSITE (1)
THRUST CHAMBERS	SAME AS STRIPPED GEMINI		113	114
INSTALLATION – THRUSTERS	SAME AS STRIPPED GEMINI		8	18
PRESSURANT	REFER TO SECTION 7.2		14	68
PRESSURANT TANKS	SAME AS STRIPPED GEMINI		155	
	1 @ 25.26 IN. I.D. SPHERICAL, GEMINI DESIGN CRITERIA, 1.066 NON-OPTIMUM FACTOR			102
PRESSURANT VALVES	MODIFIED GEMINI COMPONENTS		16	14
PROPELLANT (2)	REFER TO SECTION 7.2		3643	884
OXIDIZER TANKS	SAME AS STRIPPED GEMINI		54	
	1 @ 37.00 IN. I.D. HEMISPHERICAL AND WITH 2.1 IN. CYLINDRICAL SECTION, GEMINI DESIGN CRITERIA, 1.21 NON-OPTIMUM FACTOR			39
FUEL TANK	SAME AS STRIPPED GEMINI		41	
	SAME AS COMPOSITE SYSTEM OXIDIZER TANK			39
OXIDIZER VALVES	MODIFIED GEMINI COMPONENTS		2	2
FUEL VALVES	MODIFIED GEMINI COMPONENTS		2	2
MISCELLANEOUS VALVES	SHUT-OFF VALVES AND/OR TUBING CUTTERS		1	7
LINES AND MISCELLANEOUS	NUMBER OF THRUSTERS AND LENGTH OF LINES		5	27
ELECTRICAL	NUMBER AND SIZE OF THRUSTERS		10	50
SUPPORT STRUCTURE	STATICAL EQUATION FOR TUBULAR TRUSS WEIGHT. SUPPORT WT. = (.875)(WEIGHT SUPPORTED)$^{.65}$		191	141
TOTAL (2)		5762	4255	1507

(1) THE OAMS, STATION STABILIZATION AND STATION MAINTENANCE ARE IN ONE SYSTEM

(2) DOES NOT INCLUDE STATION KEEPING PROPELLANT 1234 LB.

11.1.13 (Continued)

a specifically designed composite system that performs the functions of the two separate systems in Stripped Gemini.

Propulsion system weight derivations are based on Gemini component weights, extrapolations of Gemini data and comparison with other known data points.

Where new tankage is necessary or desirable, weights are determined by adjusting theoretical weights by the analytical-to-actual (non-optimum) factors presented in Figure 11.1-1. These factors are based primarily on Gemini pressure vessel data.

GEMINI TANKAGE NON-OPTIMUM FACTORS

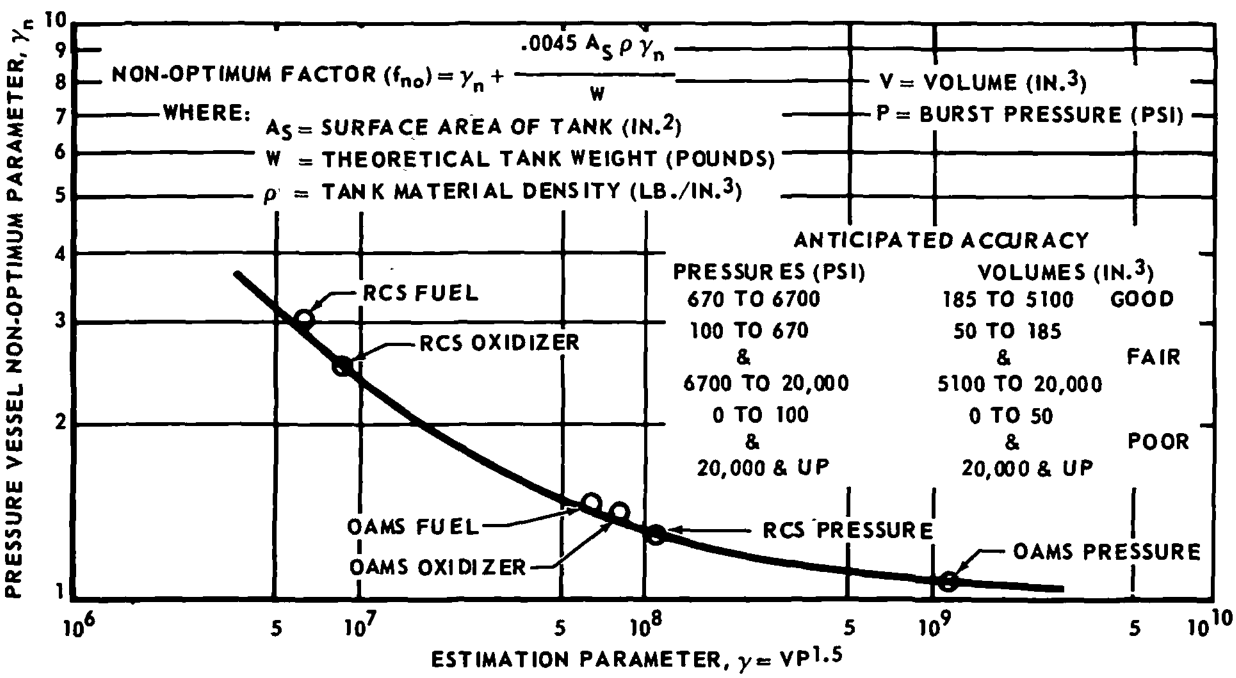


FIGURE 11.1-1

Propulsion system support structure is obtained with the empirical equation reflected in Figure 11.1-2.

11.1.14 Launch Escape System - The Saturn-launched Ferry/Supply Spacecraft utilizes a new escape system. Gemini is modified to accept an escape tower

SUPPORT (TUBULAR TRUSS) WEIGHT vs EQUIPMENT SUPPORTED WEIGHT

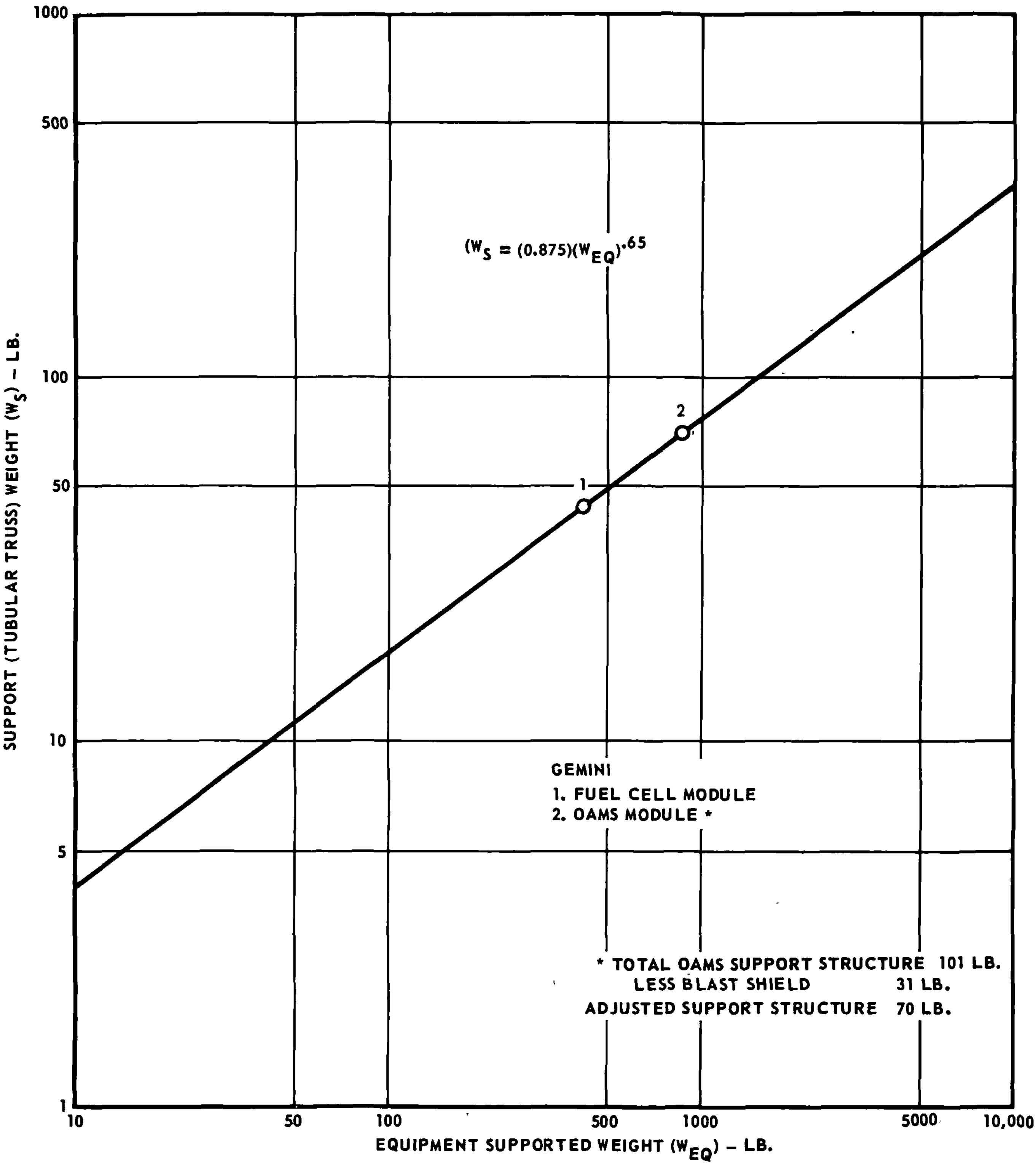


FIGURE 11.1-2

11.1.14 (Continued)

configuration similar to the Mercury design. The modifications are a combination of structural beef-ups and geometry revisions. The net effects are a 29-pound structural weight increase,and an "effective" weight increase of 12.5 percent of escape system weight. A weight breakdown of the system is contained in Table 11.1-24.

TABLE 11.1-24
LAUNCH ESCAPE SYSTEM
FERRY/SUPPLY SPACECRAFT

ITEM	WEIGHT LB.
ESCAPE ROCKET MOTOR	924
TOWER JETTISON MOTOR	26
SEPARATION PROVISIONS	77
TOWER STRUCTURE	600
BALLAST (ABORT C.G. CONTROL)	460
TOTAL – LAUNCH ESCAPE SYSTEM	(2087)

11.1.15 Re-entry Center of Gravity - Offset Ballast - The lateral range requirements and associated center of gravity limits are considered to be similar to those of Gemini. The center of gravity envelope for a re-entry from a 161 na. mi. orbit is presented in Figure 11.1-3. The limits (from 40 to 50 na. mi. range) which are based on the Gemini mission profile and the afterbody shingle heating limit, respectively, are assumed to be applicable for a Ferry Spacecraft re-entry. The locations of the center of gravity for Gemini and the three Ferry Spacecraft are shown on Figure 11.1-3. The nose dock-nose moor version does not require ballast. The nose dock-side moor version, however, due to addition of a circular auxiliary hatch in the present hatch, and side mooring fitting requires 23 pounds of ballast. Due to similar factors, the aft dock-aft moor version needs 22 pounds of ballast.

11.2 Weight Aspects of Various Alternates - Weight aspects of various alternate systems are presented in this section. These systems can be divided into the following categories:

GEMINI OFFSET ENVELOPE

RE-ENTRY FROM 161 NA.MI. CIRCULAR ORBIT
OPEN LOOP BANK ANGLE – 45°
NOMINAL ARDC ATMOSPHERE
4 RETROS, $\beta = -16°$

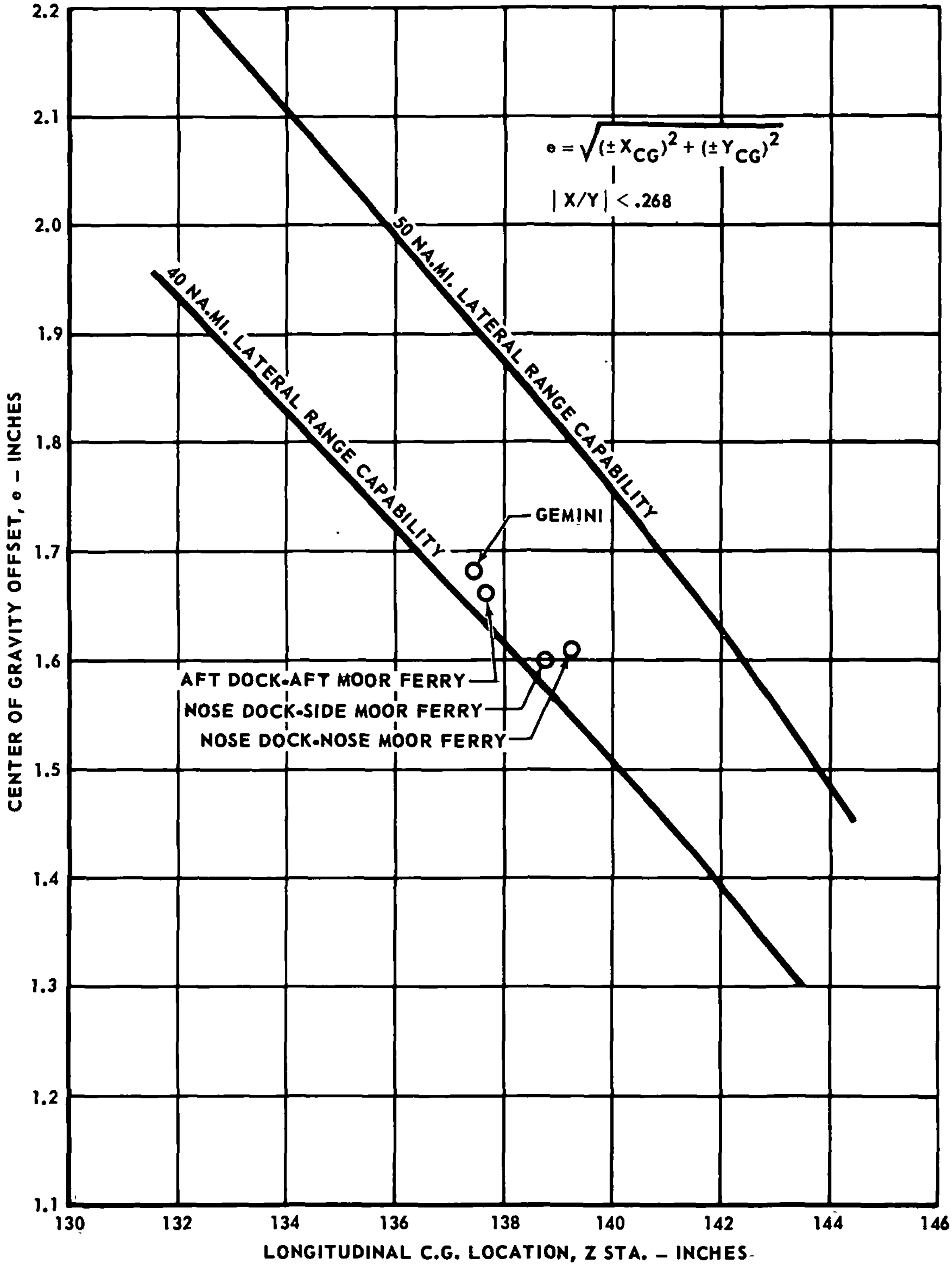


FIGURE 11.1-3

11.2 (Continued)

A. Alternates that increase spacecraft weight (Table 11.2-1)

B. Alternates that reduce spacecraft weight (Table 11.2-2)

C. Miscellaneous Alternates (Table 11.2-3)

Most of the alternates affect the weights of more than one of the systems discussed in Section 11.1.

11.3 Weight Margin/Weight Growth - Adequate weight margins are required on preliminary designs to insure that launch vehicle capabilities are not exceeded due to weight growth.

Spacecraft weight growth may be categorized as follows:

A. Weight growth due to capability increases, i.e., changes in basic ground rules or design criteria.

B. Weight growth due to technological reasons, e.g., results of laboratory tests, more thorough systems analyses.

Results of an analysis of the weight growth history of McDonnell products are shown in Figure 11.3-1. Based on this analysis, it is believed that the weight growth of a proposed spacecraft, excluding capability increases, should not exceed eight percent, with a probability of 80 to 85 percent.

As the design progresses, the "adequate" margin will decrease, due to increased confidence based on calculated and actual weights. The adequate margin for a spacecraft at any phase of design can be evaluated, based on the percent of total weight in each of four categories. These are:

A. Estimated items

B. Preliminary calculations

C. Released engineering

D. Actual hardware

Weighting factors are assigned to each of these categories based on past McDonnell

TABLE 11.2-1

ALTERNATE VERSIONS THAT INCREASE SPACECRAFT WEIGHT

ITEM	WEIGHT INCREASE LB.	EFFECTIVITY	NOTES
INCREASE POST-SEPARATION ORBITING FROM 3 TO 24 HR. AND REDUCE POST-LANDING FROM 36 HR. TO 4 HR.	115	FERRY AND FERRY/SUPPLY SPACECRAFT	INCLUDES INCREMENTS FOR BREATHING OXYGEN, FOOD AND ADDITIONAL POWER.
INCREASE CATCH-UP TIME FROM 19.9 HR. TO 43.9 HR. (NORMAL GEMINI).	32	FERRY AND FERRY/SUPPLY SPACECRAFT	REQUIRES REPLACEMENT OF LOADED FOOD, BREATHING OXYGEN AND PORTION OF FUEL CELL REACTANTS.
CHANGE LANDING SYSTEM CHANGE TO PARAGLIDER CHANGE TO PARASAIL	 731 498	FERRY AND FERRY/SUPPLY SPACECRAFT	BASIC VERSION IS 84' DIA. GEMINI PARACHUTE. WEIGHT INCREASE IS INCREMENT OBTAINED BY DELETING PARACHUTE AND ADDING SYSTEM SHOWN.
NOSE EGREES INSTEAD OF EGRESS TO SPACE.	38	FERRY AND FERRY/SUPPLY SPACECRAFT NOSE DOCK–NOSE MOOR VERSIONS	PRESSURIZED TUNNEL ADDED AND PORTABLE OXYGEN SUPPLIES DELETED.
UTILIZE SAME INSTRUMENTATION AS GEMINI INSTEAD OF DELETING R & D ITEMS.	67	FERRY AND FERRY/SUPPLY SPACECRAFT	
CHANGE "REFUSE" RETRO ΔV FROM 10 FPS TO 300 FPS (UNMANNED SUPPLY)	345	UNMANNED SUPPLY SPACECRAFT	REQUIRED ONLY FOR IMMEDIATE ELIMINATION OF SPACE DEBRIS.
REPLACE BATTERIES WITH GEMINI FUEL CELLS FOR UNMANNED SUPPLY.	 67 110	UNMANNED SUPPLY SPACECRAFT: STRIPPED GEMINI SPECIFIC DESIGN	COMPARISON CONSIDERS BASIC SYSTEMS PLUS MOUNTING AND INSTALLATION. GEMINI FUEL CELL TANK SIZE IS REDUCED FOR THE REDUCED REACTANT REQUIREMENT.
REPLACE FUEL CELLS WITH GEMINI TYPE BATTERIES.	81	FERRY AND FERRY/SUPPLY SPACECRAFT	CONSIDERS SYSTEMS PLUS MOUNTING AND INSTALLATION BUT FUEL CELL REACTANT CONTAINERS ARE NOT REDUCED.
RETROGRADE USING HOHMANN TRANSFER FROM 250 NA.MI. TO 160 NA.MI. FOLLOWED BY NORMAL GEMINI RE-ENTRY IN LIEU OF BASIC VERSION.	101	FERRY AND FERRY/SUPPLY SPACECRAFT	BASIC VERSION CONSISTS OF ADDING TWO ADDITIONAL ROCKETS TO THE GEMINI SYSTEM AND 18 LB. OF ABLATION MATERIAL TO THE HEAT SHIELD. FIVE OR SIX ROCKETS ARE FIRED, THE SIXTH ROCKET BEING PROVIDED FOR REDUNDANCY.
USE FINNED ADAPTER LAUNCH ESCAPE SYSTEM INSTEAD OF TOWER.	137	FERRY/SUPPLY	PRELIMINARY RESULTS BASED ON EFFECTIVE WEIGHTS.

TABLE 11.2-2

ALTERNATE VERSIONS THAT REDUCE SPACECRAFT WEIGHT

ITEM	WEIGHT RE-DUCTION LB.	EFFECTIVITY	NOTES
DELETE RADAR TRANSPONDER SYSTEM.	40	FERRY AND FERRY SUPPLY	REDUCES GUIDANCE AND CONTROL ELECTRONICS RELIABILITY FROM .9998 TO .99578.
UTILIZE STRUT INSTEAD OF NOSE BEEF-UP FOR NOSE DOCKING.	15	NOSE MOOR VERSIONS	SPACE STATION IS PENALIZED BY STRUT WEIGHT (25 LB.) BUT FERRY WEIGHT IS REDUCED BY WEIGHT OF NOSE BEEF-UP STRUCTURE (15 LB.).
FIRE 6 OF 6 RETROROCKETS AND CONTROL L/D.	18	FERRY AND FERRY SUPPLY	WITH THESE RESTRICTIONS, NO ADDITIONAL ABLATION MATERIAL IS REQUIRED ON GEMINI HEAT SHIELD. MAXIMUM LIFT RE-ENTRY CANNOT BE MADE IF A ROCKET FAILS.
DELETE EMERGENCY DEPARTURE CAPABILITY.	0	FERRY AND FERRY SUPPLY	APPARENTLY, ALL REQUIRED FUNCTIONS CAN BE PERFORMED WITHIN THE TIME REQUIRED FOR ASTRONAUTS TO ENTER THE SPACECRAFT. THERE IS NO WEIGHT PENALTY ASSOCIATED WITH THIS CAPABILITY.
REPLACE GEMINI BATTERIES WITH AGENA BATTERIES FOR UNMANNED SUPPLY POWER.	2 14	STRIPPED GEMINI (US-1) SPECIFIC DESIGN (US-2)	US-1 AND US-2 REQUIRED 9 AND 7 GEMINI BATTERIES RESPECTIVELY. IF AGENA BATTERIES ARE USED, THE QUANTITIES REQUIRED ARE REDUCED TO 5 AND 7. COMPARISON INCLUDES BATTERIES PLUS MOUNTS AND INSTALLATION PLUS ONE BATTERY FOR REDUNDANCY IN ALL CASES.
REDESIGN CARGO ADAPTER SPECIFICALLY FOR SATURN I LAUNCH.	300	FERRY SUPPLY	BASIC CARGO ADAPTER IS DESIGNED FOR EITHER SATURN I OR IB LAUNCH.
REDESIGN FUEL CELL REACTANT CONTAINERS TO REQUIRED VOLUME IN LIEU OF USING EXISTING GEMINI CONTAINERS.	29	FERRY AND FERRY SUPPLY	BASIC VERSIONS OFF LOAD REACTANTS BUT DO NOT REDUCE CONTAINER SIZE.

TABLE 11.2-3

MISCELLANEOUS ALTERNATE VERSIONS

ITEM	WEIGHT LB.	EFFECTIVITY	NOTES
ADD SELF INJECT SYSTEM. USEFUL PAYLOAD INCREASE IN 250 NA.MI. ORBIT USEFUL PAYLOAD INCREASE IN 200 NA.MI. ORBIT	 708 680	F, US	SELF INJECT IS BASED ON A 2200 LB. THRUST AEROJET APOLLO SUBSCALE ENGINE AND UTILIZES 3603 LB. OF PROPELLANT.
PAYLOAD CAPABILITY AND/OR WEIGHT MARGIN INCREASE WITH REDUCTION IN ORBIT ALTITUDE CHANGE FROM 250 TO 210 NA.MI. ORBIT CHANGE FROM 250 TO 160 NA.MI. ORBIT	 292 610	F, US	ONE RETROROCKET IS REMOVED AT 210 NA.MI. AND ANOTHER AT 160 NA.MI. VARIATIONS IN BREATHING OXYGEN, HEAT SHIELD ABLATION MATERIAL, AND LAUNCH VEHICLE CAPABILITY ARE ALSO CONSIDERED.
PAYLOAD CAPABILITY AND/OR WEIGHT MARGIN INCREASE USING NORMAL INSTEAD OF MINUS 3 SIGMA GLV LAUNCH CAPABILITY.	500	F, US	

F = FERRY VERSIONS
US = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI & SPECIFIC DESIGN

WEIGHT GROWTH HISTORY OF MCDONNELL VEHICLES

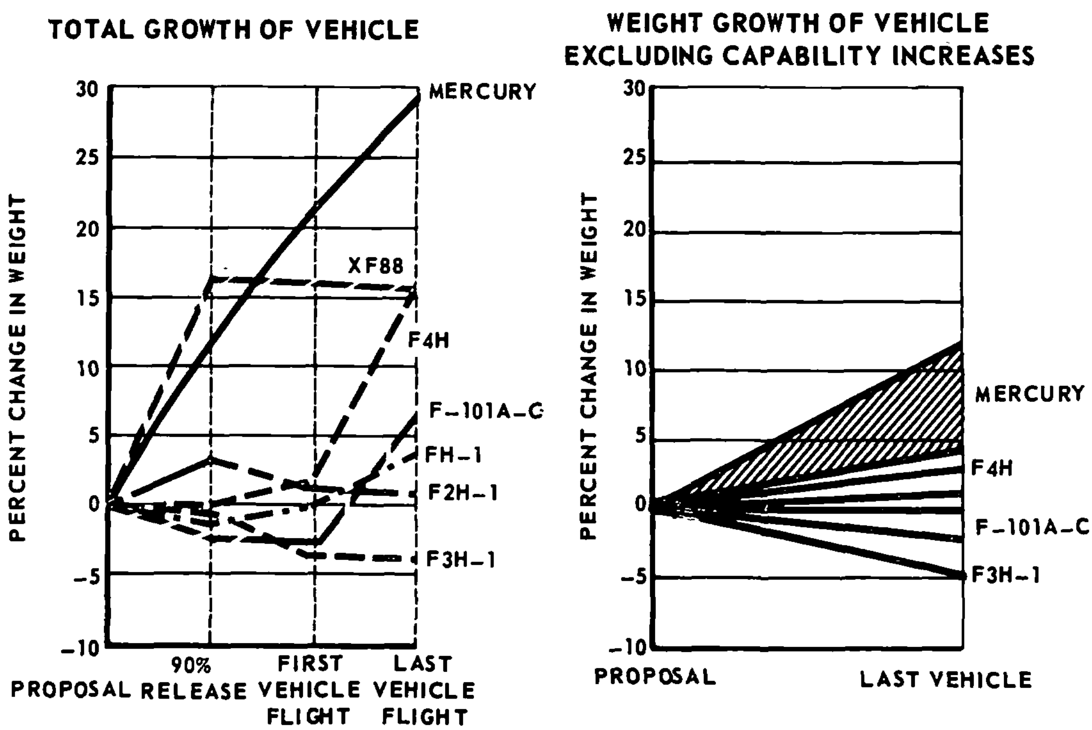
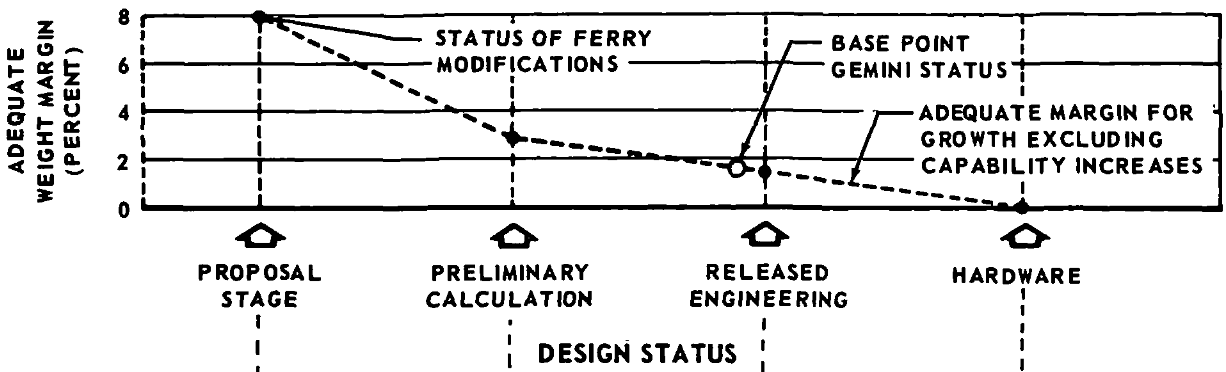


FIGURE 11.3-1

11.3 (Continued)

experience. The resulting adequate weight margin calculation procedure is shown in Figure 11.3-2.

FERRY SPACECRAFT ADEQUATE WEIGHT MARGINS



$$\text{ADEQUATE MARGIN} = (.08)\begin{pmatrix}\text{ESTIMATED}\\ \text{HARDWARE}\end{pmatrix} + (.03)\begin{pmatrix}\text{PRELIMINARY}\\ \text{HARDWARE}\end{pmatrix} + (.015)\begin{pmatrix}\text{RELEASED}\\ \text{HARDWARE}\end{pmatrix} + (.001)\begin{pmatrix}\text{ACTUAL}\\ \text{HARDWARE}\end{pmatrix} = \begin{pmatrix}\text{MARGIN}\\ \text{WEIGHT}\end{pmatrix}$$

$$\text{FOR GEMINI} = (.08)(209) + (.03)(209) + (.015)(5987) + (.001)(557) = 114 \text{ LB.}$$

$$\text{FOR NOSE DOCK NOSE MOOR FERRY} = (.08)(567) + (.03)(252) + (.015)(6023) + (.001)(413) = 145 \text{ LB.}$$

FIGURE 11.3-2

Due to the near-hardware status of Gemini, the variations considered in this study do not require "proposal stage" weight margins. Instead, adequate margins are calculated using the procedures of Figure 11.3-2 with the category totals appropriately modified to include the changes to Gemini. The adequate margin derivation for the nose dock-nose moor Ferry Spacecraft is shown in Figure 11.3-2. Similar derivations were conducted for each version considered.

Adequate margins for each of the versions are compared with available margins in Table 11.3-1. Based on this comparison, the Ferry Spacecraft margins are slightly inadequate. The margin deficiencies for the two nose docking versions, however, are quite small, i.e. 12 and 21 pounds, respectively. In addition, the required margins include a high probability (80-85 percent) of meeting or bettering

11.3 (Continued)

TABLE 11.3-1
MARGIN COMPARISON – POUNDS
FERRY SPACECRAFT

SPACECRAFT	REQUIRED MARGIN (FOR NORMAL GROWTH)	MARGIN AVAILABLE	AVAILABLE / REQUIRED (PERCENT)
GLV-LAUNCHED			
F-1	145	133	92
F-2	144	124	86
F-3	165	-147	-
SATURN I LAUNCHED			
FS-1	413	3417	830
FS-2	414	3408	825
FS-3	438	3358	765
SATURN IB LAUNCHED			
FS-1	446	4889	1090
FS-2	447	4880	1090
FS-3	472	4830	1020
GLV LAUNCHED UNMANNED SUPPLY WITH SELF-INJECT			
US-1	290	1187	410
US-2	292	656	225

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

the proposed weights. For this reason it is felt that with careful weight control and monitoring of all design changes, these two versions could be built to the study ground rules and within the -3 sigma launch vehicle capability.

Since the margin deficiency for the aft docking version is somewhat larger (314 lb.) this version may require relaxation of some of the study ground rules. A slight relaxation in ground rules can significantly improve the weight margin picture. For example, if station orbit altitude is reduced from 250 to 210 na. mi., the available margin is increased by 292 pounds. A comparison of adequate and available margin versus altitude is shown in Figure 11.3-3.

Weight margins for the Ferry/Supply and Unmanned Supply Spacecraft are large in all cases, as shown in Table 11.3-1. The available margins shown result from

11.3 (Continued)

MARGIN vs ORBIT ALTITUDE

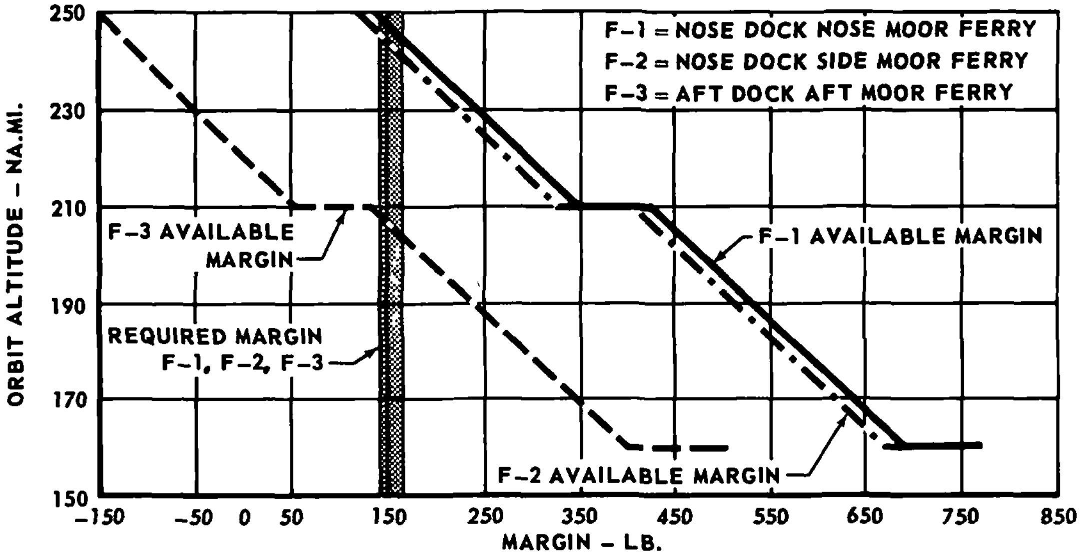


FIGURE 11.3-3

the fact that the minimum number of supply launches needed per year, provides a greater amount of cargo capability than the logistics supply requirements for one year of MORL operation.

11.4 Logistics Requirements - A MORL logistics requirement of 45 pounds per station/day is derived in Section 4.3. The logistic requirements summary in Table 4.3-1 of Section 4.3 is used in sizing the various supply spacecrafts. The cargo capability in space station days for the supply spacecrafts is as follows:

Stripped Gemini	83 days
Specific Design	110 days
Ferry/Supply	330 days

These values are based on available volumes. Spacecraft cargo loadings are further limited by launch vehicle capabilities.

The required cargo loadings for individual spacecraft vary depending on the number of supply launches and the amount of cargo launched with the space station. The weight margins for the Ferry/Supply and Unmanned Supply spacecraft vary accordingly.

11.4 (Continued)

If 30 days supplies are launched with the space station, the following minimum number of launches will be required for the particular supply versions:

A. Saturn IB launched Ferry/Supply - 1 launch per year

B. Saturn I launched Ferry/Supply - 2 launches per year

C. Specific Design-Unmanned Supply - 3 launches per year

D. Stripped Gemini Unmanned Supply - 4 launches per year

Cargo breakdowns and weight margins based on these assumptions are presented in Table 11.4-1.

TABLE 11.4-1

CARGO/GROWTH OFFSET CAPABILITY – POUNDS

SPACECRAFT	FS–1		FS–2		FS–3		US–1	US–2
LAUNCH VEHICLE	SI	SIB	SI	SIB	SI	SIB	GLV WITH SELF INJECT	
CARGO LOGISTICS SUPPLIES – DAYS	165	330	165	330	165	330	83	110
TOTAL	(10,295)	(18,828)	(10,286)	(18,819)	(10,236)	(18,769)	(4639)	(5255)
STATION KEEPING PROPELLANT	1838	3859	1838	3859	1838	3859	912	1234
CARGO IN PRESSURIZED AREA	2000	4000	2000	4000	2000	4000	1010	1335
CARGO IN UNPRESSURIZED AREA	3040	6080	3040	6080	3040	6080	1530	2030
GROWTH/MARGIN	3417	4889	3408	4880	3358	4830	1187	656

* SYSTEM WEIGHT INCLUDED IN BASIC VEHICLE
FS–1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS–2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS–3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US–1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US–2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN
SI = SATURN I
SIB = SATURN IB

12. INTERFACES

12.1 Ferry/MORL Interfaces - During rendezvous, docking, transfer of crew and cargo, standby and departure of the Ferry and supply vehicles, compatibility requirements for mechanical and electrical interfaces impose constraints on both the MORL and the Ferry Spacecraft.

12.1.1 Control of Rendezvous - Rendezvous is accomplished by techniques similar to Gemini. Manned rendezvous is controlled by the astronauts in the Ferry Spacecraft and unmanned rendezvous is controlled by astronauts onboard MORL. Manned spacecraft are modified to permit the use of MORL systems as a back-up. The guidance and communication interfaces that implement this system, and their functional operation, are shown in Figures 7.5-2 and 7.5-7. The specifications assumed for the MORL communications system are listed in Figure 7.5-3. Circularly polarized antennas are desirable on MORL to insure adequate signal reception regardless of the relative orientation of transmitting and receiving antennas. The PCM/FM telemetry receiver shown on MORL is considered desirable to monitor the progress of unmanned rendezvous missions.

Trade-offs exist with respect to those links where the maximum range is less than the 250 na. mi. design range for Ferry-MORL communications. Directional antennas can be installed on MORL, or transmitter power can be increased to achieve design range. The last column of Figure 7.5-3 specifies the size of the antenna for the increase in gain in db required.

12.1.2 Docking Procedures and Mechanisms - Docking procedures for nose docked Ferry Spacecraft are identical to those for Gemini. Requirements on the MORL Systems consist of stabilization of the station during docking and provisions of a modified Agena docking cone. Three such docking stations are considered to be located on the MORL. Nose dock-side moor configurations require that these docking cones have the ability to rotate the spacecraft to the side moored position.

12.1.2 (Continued)

The rotation will be controlled from inside the MORL or remotely from the spacecraft. Since side moored concepts include provisions for pressurized crew transfer three entry ports into the MORL should be provided. Aft docked Ferry Spacecraft are docked on the longitudinal axis of the MORL (forward end). Repositioning to the side of MORL by mechanical means located on, and controlled from, the station may be desirable. Docking is accomplished without assistance from MORL. Docking and mooring methods are illustrated in Figure 2.4-1.

Unmanned Supply Spacecraft docking is performed using a single Gemini target docking cone located on the longitudinal axis of the MORL. A pressure sealing surface and access to the MORL must be provided inside the cone for pressurized cargo transfer. Docking of unmanned supply spacecraft is controlled remotely from the MORL.

Except for the aft dock-aft moor Ferry/Supply Spacecraft, which is controlled from an aft docking station located aboard the spacecraft, docking of Ferry/Supply Spacecraft is accomplished with control available to crewmen within the MORL. These concepts are illustrated in Figure 2.5-1.

12.1.3 Transfer of Personnel, Equipment and Supplies - Personnel transfer does not require physical aid from crewmen within the MORL, but external provision for ingress (e.g., airlock pressurization and hatch latching) must necessarily be provided by the space station. Crew transfer procedures and requirements are illustrated and discussed in Section 7.4.

Equipment and supply transfer is accomplished through a single cargo port located on the longitudinal axis of the MORL. Solid cargo is manually transferred from the pressurized supply compartment through a tunnel directly into the MORL. Electrical connections and umbilicals for propellants, fluids, and gases are connected and secured from within the pressurized cargo tunnel. This requires that all

12.1.3 (Continued)

supply and electrical umbilicals in the MORL be routed through the supply port.

12.1.4 Checkout - Ferry spacecraft checkout procedures involve three levels of detail: (1) continuous monitoring in the MORL of a few critical Ferry Spacecraft parameters, (2) periodic monitoring of Ferry Spacecraft instrumentation transducers utilizing the telemetry transmitter and a radio frequency coax link with the MORL, and (3) a detailed checkout of any system suspected of malfunction using the digital command system to remotely turn on and operate the suspect system and the telemetry link to acquire data for diagnosis. Appropriate displays and controls are assumed to be provided on the MORL. A patch panel and data display are required to select the items of interest from sequentially telemetered data in the second and third levels of checkout. The MORL/spacecraft monitor and checkout interface is discussed in Section 9.5.2.

12.1.5 Malfunction Corrections - These are limited to corrections that can be accomplished by relatively simple adjustment and calibration or by a change in environment, e.g., raising or lowering ambient temperature or venting excess pressures. In all other cases malfunction corrections are achieved through the use of redundant components. No special provisions have been made for in-orbit repair or replacement of Ferry Spacecraft equipment.

12.1.6 Environmental Control Interface - The station will supply oxygen for breathing, pressurization and ventilation of the astronauts during part of the crew transfer procedure. The Environmental Control System (ECS) lines and high pressure oxygen supply will be stowed on MORL after transfer (see Section 7.4.1). Pressurizing gas is supplied to the Ferry Spacecraft during standby to prevent potential cold welding. The station atmosphere is used to pressurize cargo compartments. Oxygen and nitrogen are supplied to the MORL as 100 psi gases directly from tanks in the supply spacecraft. Electrical power from MORL operates the heaters in the

12.1.6 (Continued)

cryogenic supply tanks.

During standby, electrical power from MORL is needed for heaters on components in the Ferry and Supply Spacecraft. Alternate thermal protection schemes which feature, (a) angled mooring and less electrical power or (b) external pressurized covers or bags carried by the MORL are discussed in Section 9.3.

12.2 Spacecraft/Launch Vehicle Interface - Mating of the spacecraft with the launch vehicle requires special adapters, in some cases, to provide mechanical clearances or matching of unequal dimensions, and provisions for electrical integration of the vehicles.

12.2.1 Ferry/Gemini Launch Vehicle (GLV) - The well established interface for the Gemini/GLV applies without change to the Ferry/GLV. This includes both mating and system integration procedures.

12.2.2 Ferry-Supply/Saturn Launch Vehicle - The Ferry/Supply Spacecraft can be launched by either the Saturn I or Saturn IB. Electrical interfaces are similar for both launch vehicles. Ferry Spacecraft instrumentation includes 54 electrical leads in the interface connection with the GLV for such functions as guidance signals, staging indications and malfunction detection system signals, including excessive attitude rates, pressure failures, temperatures and abort signals. Since lack of data on the Saturn Malfunction Detection System and other instrumentation functions prevented complete evaluation of the electrical interface, it is assumed that the GLV electrical interface applies.

Mechanical interface of the Ferry/Supply and Saturn I and IB launch vehicles involves matching of the diameter of the two vehicles and clearance of the Saturn Instrumentation Module. The diameter of the Ferry/Supply Module at the base is 154" which matches the diameter of the Saturn Instrumentation Module on the SIV stage of the Saturn I. An additional short cylindrical adapter will have to be

12.2.2 (Continued)

used to provide clearance between the Supply Module and Instrumentation Module equipment. It is expected that a hatch will be provided in the adapter to permit access to the Saturn instrumentation for on-pad maintenance. At staging, the Ferry/Supply Spacecraft will be separated from the adapter by standard Gemini shaped charges and pyrotechnic devices.

The Saturn IB instrumentation module is 260" in diameter. Mating with the 154" diameter base of the Ferry/Supply can be satisfied by use of the Apollo adapter. At staging, the Ferry/Supply Spacecraft will be separated from the adapter in the same manner as for Saturn I.

12.2.3 Unmanned Supply/GLV - Electrical interface between the Unmanned Supply Spacecraft and the GLV does not lead to redesign on the launch vehicle. The same umbilical connection which is used for the Gemini/GLV interface can be employed, although fewer signals are needed with supply spacecraft due to deletion of the cockpit displays.

The diameters of the launch vehicle upper stage and the base of the supply spacecraft are the same. However, the Unmanned Supply Spacecraft self-inject engine skirt extends beyond the base of the spacecraft. An adapter, 120" diameter by 48" in length, is added to the GLV to provide clearance between the self-inject engine and the tank of the upper stage. At spacecraft staging, the adapter stays with the launch vehicle.

12.3 Ferry/Ground Interface - The Ground Operation Support System (GOSS requirements for the MORL Ferry operations are the same as those for Gemini, with the exception of the facilities and ground equipment modifications discussed in the following paragraphs.

12.3.1 Launch Complexes - Logistic requirements for launch of Ferry and Unmanned Supply Spacecraft require additional Gemini Launch Vehicle (GLV) launch pads.

12.3.1 (Continued)

Modification of AMR Titan II launch complexes to the present launch complex 19 configuration would provide this additional capability. Basically, the modifications consist of the conversion of launch pad and blockhouse to accommodate the GLV and additional facilities for the Gemini configuration payload. Based on previous conversion of launch complex 19, Tables 12.3-1 and 12.3-2 list the required modifications and Figure 12.3-1 presents a representative schedule for modification of the launch complexes.

TABLE 12.3-1
PAD 19 CONVERSION TITAN I TO TITAN II

1. TANK FARM FUEL
2. TANK FARM (TANKS FURNISHED BY MARTIN)
3. OXIDIZER HOLDING
4. DECONTAMINATION BLDG.
5. BOOM MODIFICATIONS TO UMBILICAL TOWER
6. VEHICLE ERECTOR, MODS FOR 2ND STAGE
7. FUEL HOLDING AREA
8. 2ND STAGE ERECTOR
9. COMPLETE VEHICLE UMBILICAL POWER
10. MOD TO EXISTING ELEVATOR
11. LAUNCH DECK BLDG.
12. WATER SYSTEM CHANGES
13. BLOCK HOUSE
14. WATER SPRAY, FLUME, & SKIMMING BASIN (500,000 GAL.)
15. WATERPROOFING DECKS, ETC.
16. TRANSFER SYSTEM FUEL & OXIDIZER (1/8 MILE, 4" STAINLESS STEEL PIPE)
17. PAVING, EXCAVATIONS, DRAINAGE, GRADING (1/2 OF) POND FILL, DEMOLISH OLD TANK FARM, NEW FENCES

TABLE 12.3-2
PAD 19 ADDITIONS FOR GLV (AFTER TITAN II MODIFICATIONS)

1. READY BLDG. FOR GEMINI
2. WHITE ROOM (INCLUDING AIR CONDITIONING)
3. PERSONNEL ELEVATOR
4. ELECTRIC WIRING (EXTERIOR)
5. MISCELLANEOUS (GEMINI)
6. MAN RATED WATER SYSTEM
7. BRIDGE CRANE } WHITE ROOM
8. JIB CRANE } WHITE ROOM
9. PAVING EXCAVATION, DRAINAGE, GRADING POND FILL, DEMOLISH OLD TANK FARM, NEW FENCES (1/2 OF TOTAL-GEMINI)

ENGINEERING FOLLOW-UP AND PROJECT CONTROL

12.3.1 (Continued)

PAD 19 CONVERSION TITAN I TO TITAN II – GLV

	1962												1963											
	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D
WORK BEGUN			▼																					
CONTRACT LET									▼															
ENGINEERING COMPLETE										▼														
BEGIN WHITE ROOM											▼													
COMPLETE WHITE ROOM																	▼							
ERECTOR TEST															▼									
B.O.D. SCHEDULED *																▼								
MARTIN BEGIN AGE INSTALLATION																		▼						
MARTIN COMPLETE AGE																				▼				

* B.O.D. BENIFICIAL OCCUPANCY DATE.

FIGURE 12.3-1

Launch complexes 34 and 37 will accommodate either the Saturn I or Saturn IB launch vehicles and can be adapted to support of the Ferry/Supply Spacecraft as a Saturn payload. Although detailed design requirements, estimated costs and conversion schedules are not available, the primary modifications to accommodate the Ferry Supply appear to be:

A. Addition of an air conditioned white room facility.

B. Addition of provisions for portable Gemini AGE, such as fluid dispensing metering units and electrical and pyrotechnic test equipment at desired spacecraft levels by use of portable work platforms or by adjustment of integral platforms.

Spacecraft levels for the Ferry/Supply when mounted on the Saturn IB will be higher than when mounted on the Saturn I. Figure 12.3-2 shows a configuration of the service tower and support facilities for the Ferry/Supply on the Saturn IB.

12.3.2 Industrial Complex - The Gemini spacecraft presently requires approximately 102 days of processing in the industrial area prior to delivery to the

FERRY/SUPPLY SPACECRAFT ON COMPLEX -37
ARRANGED FOR SATURN IB LAUNCH.

1— PIVOT MOUNTED SUPPORT RING PERMITS CRANE HOIST OF FULL S/C THRU, OR DISPLACEMENT OF ENTIRE WHITE ROOM TENT.
2— ALTERNATE SUPPORT METHOD HANG TENT MAIN RING FROM HIGHER PLATFORM. REMOVE WITH SILO ENCLOSURE.
3— ZIPPERED CANOPY PERMITS HOIST ACCESS BY QUADRANT, OR
4— FULL OPENING. ALTERNATE TYPE –
4— FLAT PLASTIC DISC CEILING
5— *SOFT PLASTIC TENT PROVIDES FOR MAINTENANCE OF WHITE-ROOM CONDITIONS
6— HALYARDS & SHEAVES MANUALLY RAISE LOWER EDGE OF W.R. TENT
7— S/C GROUND COOLING SYSTEM
8— OXYGEN SYSTEM-LEAKAGE TESTER *(52E180027)* & SUIT/CABIN PURGE
9— S/C FUEL METERING UNIT (52E420025)
10— *GEMINI REENTRY MODULE
11— PLATFORM SEGMENT RE-MOVED TO CLEAR EXCAPE UNIT.
12— GEMINI LAUNCH ESCAPE SYS. (WHITE ROOM INSTALLATION)
13— GEMINI STANDARD ADAPTER
14— S/C OXIDIZER METERING UNIT (52E420025)
15— *WHITE ROOM TENT, LOWER
16— PORTABLE PLATFORM
17— *GEMINI SUPPLY ADAPTER
18— KIT-ADAPTER-SET OF SIX HOSES REQ. FOR FLUIDS TRANSFER TO SUPPLY ADAPTER FROM UMB. ARM.

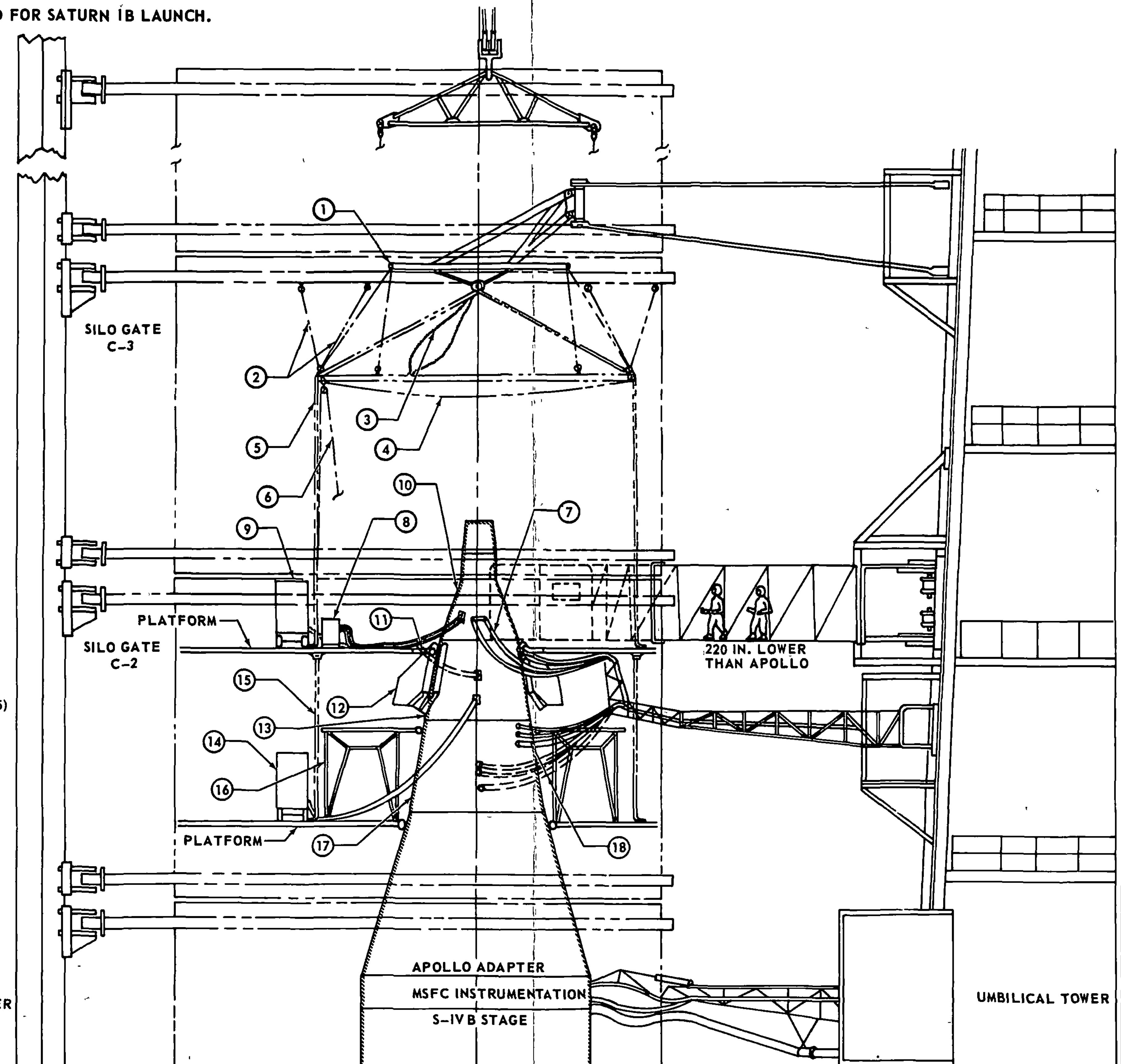


FIGURE 12.3-2

12.3.2 (Continued)

launch pad for mating to the launch vehicle. Total time in assembly, checkout, mating system test and launch countdown approximates 112 days. Following the same philosophy for pre-launch preparation of the Ferry and Supply spacecraft for the MORL, the processing time will be equivalent to Gemini for each Ferry Spacecraft.

For Ferry/Supply spacecraft, approximately the same schedule will apply, since the preparation of the supply adapter parallels the ferry assembly and checkout procedures. The assembly and checkout flow for Ferry/Supply Spacecraft is shown in Figure 12.3-3.

Unmanned Supply Spacecraft can be prepared and launched within 85 days from receipt at the launch site. The assembly and checkout flow for the Unmanned Supply Spacecraft is shown in Figure 12.3-4.

Floor area requirements in the Industrial Area for each Ferry are the same as for Gemini. The supply module and Unmanned Supply Spacecraft each require a little less floor area than the Gemini. A large portion of the required space for processing Gemini is occupied by supply, parts, shop equipment and spacecraft AGE which can be time-shared for all spacecraft in flow. Present floor space allotted to Gemini operations in the Industrial Area can accommodate about six spacecraft in flow at one time. This indicates that even with careful scheduling, concurrent programs requiring assembly and checkout of Gemini, Ferry and Supply Spacecraft will probably require an increase in allotted floor space. Expansion of white room facilities will be needed to accommodate parallel preparations of spacecraft for the MORL program.

Refurbishment of recovered Ferry Spacecraft can be accomplished in the Industrial Area or in the factory. A schedule for refurbishment of the re-entry capsule, replacement of jettisoned sections and assembly and checkout for launch, is shown in Figure 12.3-5. This turnaround procedure is based on accomplishing refurbishment at the launch site Industrial Area.

FLOW CHART
GEMINI FERRY SPACECRAFT

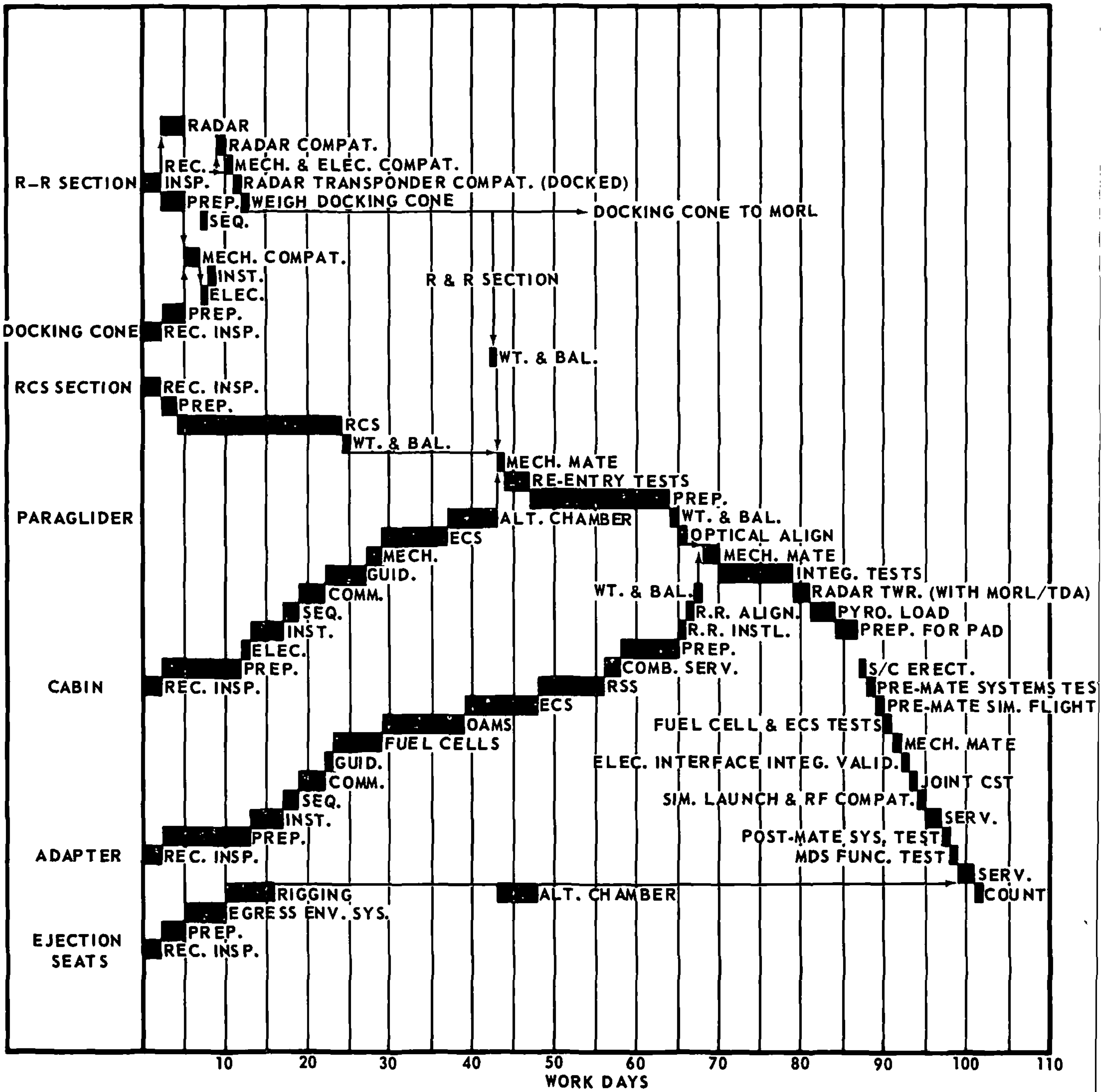


FIGURE 12.3-3

12.3.2 (Continued)

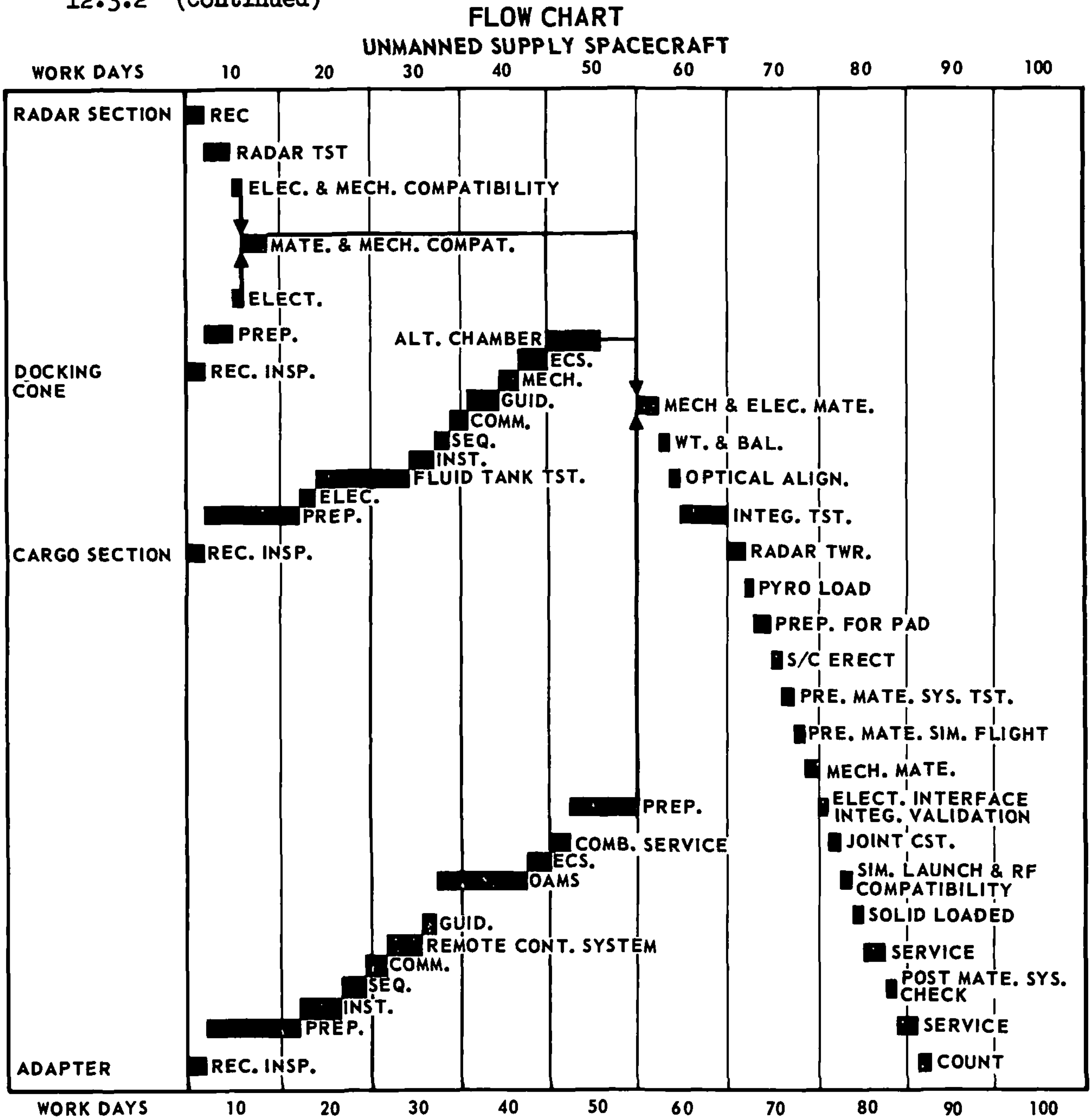


FIGURE 12.3-4

Conversion of recovered re-entry modules to Stripped Gemini Unmanned Supply Spacecraft necessitates some structural changes and relocation or deletion of subsystems. This operation should be performed in the factory where the necessary facilities, tooling and technical skills are available. Although there is insufficient data available for an accurate estimate of time required, it is expected that the conversion could be accomplished in the factory in about 3 months.

TURNAROUND

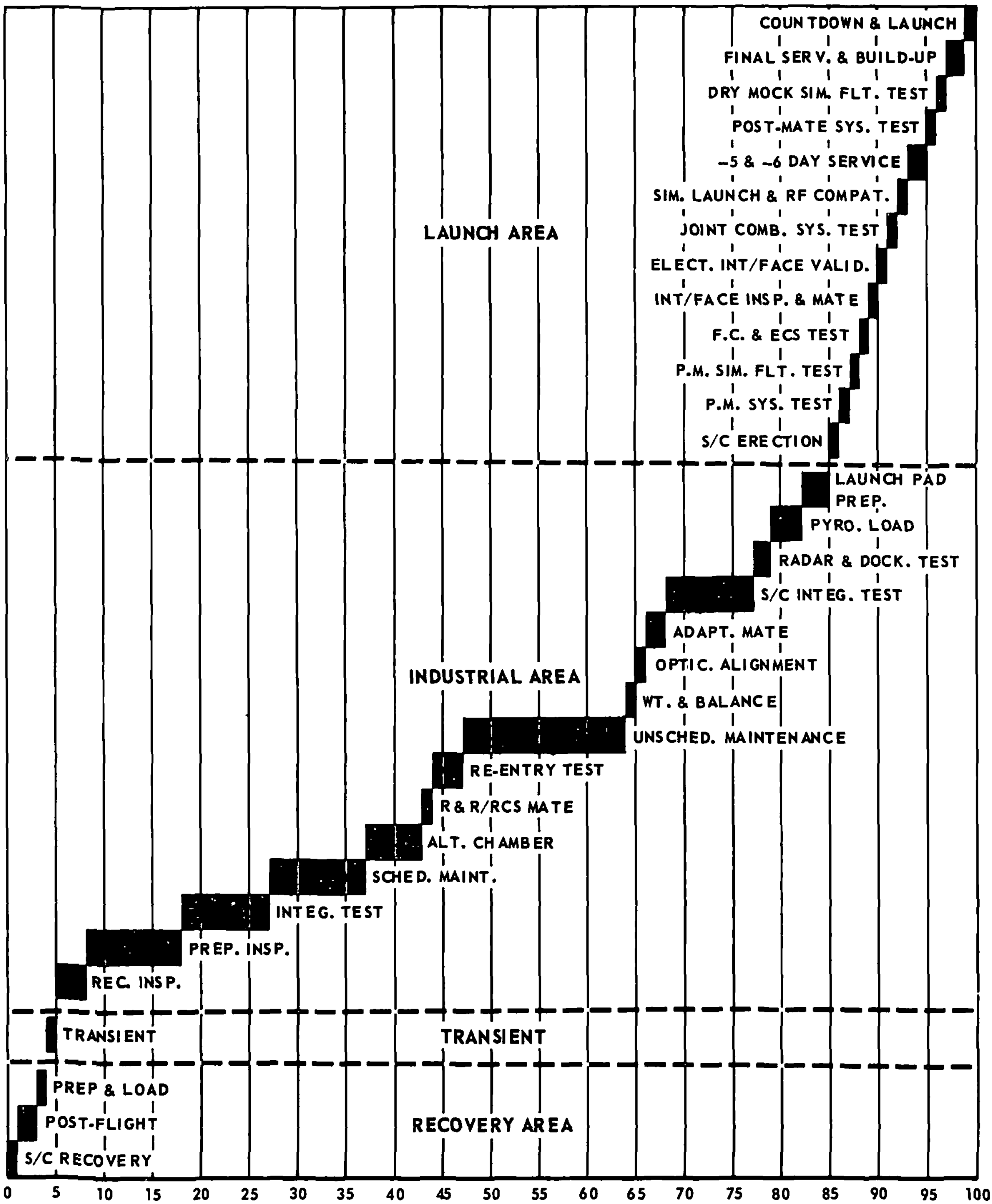


FIGURE 12.3-5

12.3.3 Aerospace Ground Equipment - There are a total of 622 items of Aerospace Ground Equipment (AGE) for support of the Gemini spacecraft. The procedures used for Gemini and the suitability to the Ferry, Ferry/Supply and Unmanned Supply Spacecraft are shown in Table 12.3-3. Gemini AGE applicability based on these procedures is presented in Table 12.3-4.

TABLE 12.3-3
GEMINI PROCEDURE APPLICATION

GEMINI ASSEMBLY, CHECKOUT AND MAINTENANCE PROCEDURES	FERRY SPACECRAFT	FERRY/SUPPLY ADAPTER	UNMANNED SUPPLY SPACECRAFT
I. INDUSTRIAL AREA			
A. SPACECRAFT ACCEPTANCE			
1. INSTRUMENT SYSTEM BENCH CHECK	X		X
2. RECEIVING & INSPECTION	X	X	X
3. VSWR CHECKS	X		X
4. MATING PROCEDURES	X	X	X
5. SOLID SUPPLY CONTAINER REMOVAL		X	X
B. RENDEZVOUS & RECOVERY SECTIONS			
1. PREPARATION PERIOD	X		X
2. ALIGNMENT & ADJUSTMENT (R & R – TDA)	X		X
3. SEQUENTIAL SYSTEM TEST	X	X	X
4. R & R SECTION LESS RADAR MODULE RETAINED AT (O & C BUILD)	X		X
5. RADAR FUNCTIONAL TEST	X		X
6. RADAR BORESIGHT (MECH)	X		
7. RADAR BORESIGHT (ELECT)	X		
8. RADAR MODULE RIST CALIBRATION	X		X
9. RADAR MODULE & TDA COMPATIBILITY TEST	X		X
C. RECOVERY CONTROL SECTION			
1. PREPARATION PERIOD (NLG & PARAWING REMOVED)	X		
2. LEAK & VALIDATION (PARAWING SYSTEM)	X		
3. LEAK & VALIDATION	X		
4. RCS & OAMS SERVICING	X		
5. RCS STATIC FIRING TEST	X		
6. RCS & OAMS SERVICING	X		
7. EQUIPMENT INSTALLATION (NLG & PARAWING)	X		
8. WEIGHT & BALANCE			
9. ASSEMBLED RCS SECTION TO O & C BLDG. FOR MATE	X		
D. CABIN OR SUPPLY SECTION			
1. PREPARATION PERIOD	X		X
2. EJECTION SEAT REMOVAL	X		
3. INSTRUMENTATION INSTALLATION	X		
4. ELECTRICAL SYSTEM TEST	X		X
5. INSTRUMENTATION SYSTEM TEST	X		X
6. SEQUENTIAL SYSTEM TEST	X		X
7. COMMUNICATION SYSTEM TEST	X		X
8. CABIN GUIDANCE CONTROL TEST	X		X
9. MECHANICAL SYSTEMS FUNCTIONAL TEST	X		X
10. LEAK AND VALIDATION TEST	X		X
11. EJECTION SEAT INSTALLATION	X		
12. ENVIRONMENTAL CONTROL SYSTEM FUNCTIONAL TEST	X		X
13. EJECTION SEAT REMOVAL	X		
14. COOLANT AND WATER MANAGEMENT SYSTEM TEST	X		X
15. CABIN OR SUPPLY SECTION TO O & C BLDG. FOR MATING	X		X
E. EJECTION SEAT			
1. RECEIVING INSPECTION	X		
2. ECS EGRESS SYSTEM FUNCTIONAL TEST	X		
3. ALTITUDE CHAMBER TEST	X		
4. INSTALLATION OF SEATS	X		
5. REMOVAL OF SEATS	X		
6. PREPARATION PERIOD	X		
7. INSTALLATION OF SEATS	X		
8. REMOVAL OF SEATS	X		
9. EJECTION SEAT BUILD-UP	X		
10. WEIGHT AND BALANCE	X		
11. STORE EJECTION SEATS UNTIL PAD INSTALLATION	X		
F. ADAPTER SECTION (EQUIPMENT)			
1. PREPARATION PERIOD	X		X
2. ADAPTER INSTRUMENTATION SYSTEM INSTALLATION	X		X
3. INSTRUMENTATION SYSTEM TEST	X		X
4. SEQUENTIAL SYSTEM TEST	X		X
5. COMMUNICATIONS SYSTEMS TEST	X		X
6. ADAPTER GUIDANCE & CONTROL SYSTEMS TEST	X		X
7. REMOTE CONTROL SYSTEMS TEST			X
8. RSS & FUEL CELL FUNCTIONAL TEST (L & V)	X		
9. RSS & FUEL CELL FUNCTIONAL TEST (GASEOUS)	X		
10. LEAK AND VALIDATION TEST	X		X
11. RCS &/OR OAMS SERVICING	X		X
12. OAMS STATIC FIRING TEST	X		X
13. RCS &/OR OAMS SERVICING	X		X
14. ECS FUNCTIONAL TEST	X		X
15. COOLANT FUNCTIONAL TEST	X		X
16. CRYOGENIC SYSTEM FUNCTION TEST	X		
17. RSS AND FUEL CELL FUNCTIONAL TEST (CRYOGENICS)	X		
18. CRYOGENIC SYSTEM SERVICING	X		
19. COMBINED HYPERGOLICS & CRYOGENICS SERVICING & OPERATION (SIMULATED PAD OPERATIONS)	X		
20. COMBINED HYPERGOLICS SERVICING AND OPERATION (SIMULATED OPERATION)			X
21. PREPARATION PERIOD	X		X
22. PYROTECHNIC BUILD-UP	X		X
23. RETROGRADE ROCKET ALIGNMENT	X		X
24. WEIGHT AND BALANCE	X		X
25. ADAPTER SECTION TO O & C BLDG. FOR MATE	X		X
G. ADAPTER SECTION (SUPPLY)			
1. PREPARATION PERIOD		X	
2. LEAK AND VALIDATION (L & V) TEST (FLUID TANKAGE)		X	
3. OAMS SYSTEM SERVICING		X	
4. OAMS SYSTEM STATIC FIRING TEST		X	
5. COMBINED FLUID TANKAGE SERVICING AND OPERATION (SIMULATED PAD OPERATION)		X	
6. PYROTECHNIC BUILD-UP		X	
7. WEIGHT AND BALANCE		X	
8. ADAPTER SECTION TO O & C BUILDING FOR MATE		X	
9. S/C-ADAPTER INTEGRATED TEST		X	
10. DEMATE S/C AND ADAPTER SECTION		X	
11. PREPARE FOR MOVE TO LAUNCH PAD		X	
H. PARAWING			
1. PARAWING RECEIVING & INSPECTION	X		
2. PARAWING PRESSURE TEST & PACKING	X		
3. WEIGHT AND BALANCE FOR TOTAL GROSS WEIGHT	X		
4. STORE UNTIL INSTALLATION ON PAD			
I. TDA SECTION			
1. REMOVE TRANSPONDER FROM TDA MOVE BOTH TO RADAR RANGE	X		
2. COMMUNICATIONS SYSTEMS TEST	X		X
3. REINSTALL TRANSPONDER IN TDA MOVE TO O & C BLDG.	X		
4. ELECTRICAL SYSTEMS TEST	X		X
5. INSTRUMENTATION SYSTEMS TEST	X		X
6. FLIGHT READY TDA TO RADAR RANGE	X		
7. RADAR/TRANSPONDER COMPATIBILITY TEST	X		X
8. INSTALL RADAR MODULE IN R & R SECTION	X		X
J. R & R AND TDA SECTION			
1. TDA & R & R SECTION COMPATIBILITY	X		X
2. RADAR & TRANSPONDER COMPATIBILITY	X		X
3. WEIGH TDA	X		X
4. WEIGHT AND BALANCE	X		X
5. RETURN TO O & C BUILDING FOR MATE	X		X
K. CABIN AND RCS SECTION			
1. MATING	X		
2. LEAK AND VALIDATION (PARAWING SYSTEM)	X		
3. LANDING SYSTEM FUNCTIONAL CHECK	X		
L. SUPPLY AND RENDEZVOUS SECTION (TDA)			
1. MATING			X
2. LEAK AND VALIDATION (LIQUID TRANSFER LINES)			X
3. VALIDATION ELECTRIC CONNECTIONS			X
M. CABIN/RCS AND R & R SECTION			
1. MATING	X		
2. RE-ENTRY CONFIDENCE TEST	X		
3. PREPARATION PERIOD	X		
4. WEIGHT AND BALANCE	X		
5. RE-ENTRY CONFIGURATION ALIGNMENT	X		
6. RETURN TO O & C BUILDING	X		
N. SPACECRAFT FINAL CONFIGURATION			
1. MATING (EQUIPMENT ADAPTER)	X	X	X
2. EJECTION SEAT INSTALLATION	X		
3. ALTITUDE CHAMBER TEST	X		X
4. EJECTION SEAT REMOVAL	X		
5. MATING (SUPPLY ADAPTER)	X	X	
6. CONFIDENCE TEST	X	X	X
7. SIMULATED FLIGHT	X	X	X
8. GEMINI/MORL COMPATIBILITY TEST	X	X	X
9. DEMATE (SUPPLY ADAPTER)		X	
10. PYROTECHNIC BUILD-UP	X	X	X
11. PAD PREPARATION	X	X	X
II. LAUNCH COMPLEX			
1. ADAPTER/LAUNCH VEHICLE COMPATIBILITY TEST, MATE & PYROTECHNIC BUILD-UP	X	X	X
2. SPACECRAFT ERECTION	X		X
3. PREMATE SYSTEMS TEST	X	X	X
4. PREMATE SIMULATED FLIGHT	X	X	X
5. FUEL CELL & ECS TEST	X		X
6. MATE MECH.	X	X	X
7. PYROTECHNIC BUILD-UP (S/C & SUPPLY ADAPTER)	X	X	X
8. ELECTRICAL INTERFACE INTEGRATED VALIDATION	X	X	X
9. JOINT CST	X	X	X
10. SIMULATED LAUNCH & RF COMPATIBILITY	X	X	X
11. SERVICE	X	X	X
12. POST MATE SYSTEM TEST	X	X	X
13. MALFUNCT. DETECT. SYS. FUNCT. TEST	X	X	X
14. SERVICE	X	X	X
15. COUNT	X	X	X

TABLE 12.3–4
QUANTITATIVE RELATIONSHIP
EXISTING GEMINI AGE APPLICATION vs NEW AGE REQUIREMENT

GEMINI AGE QUANTITY BY SYSTEM APPLICATION	GEMINI AGE APPLICABILITY			GEMINI AGE MODIFICATIONS			NEW AGE REQUIREMENT		
	F	FS	US	F	FS	US	F	FS	US
M.A.C. SPECIAL									
1. GROUND HANDLING 78	78	77	38	0	0	2	0	6	3
2. GROUND OPERATING 1	1	1	1	0	0	0	0	0	0
3. GROUND TEST 9	7	6	2	0	0	0	2	3	3
4. ALIGNMENT AND MEASUREMENT 10	10	9	2	0	1	2	0	0	1
5. BIOLOGICAL SUPPORT SYSTEM 138	138	138	76	0	0	0	0	0	0
6. COMMUNICATION SYSTEM 22	22	22	4	0	0	4	0	0	1
7. COMPLEX CABLING 16	15	15	14	1	1	1	0	0	0
8. ELECTRICAL SYSTEM 67	67	67	11	0	0	0	0	0	0
9. FACILITY SUPPORT 6	6	6	6	0	0	0	0	0	0
10. FLIGHT CONTROL 50	50	48	37	1	1	1	1	1	2
11. OPERATIONAL RECOVERY 4	4	4	0	0	0	0	0	0	0
12. PHOTOGRAPHIC SYSTEM 3	3	3	3	0	0	0	0	0	0
13. PYROTECHNIC SYSTEM 11	11	11	9	0	0	0	0	0	0
14. REACTION CONTROL SYSTEM 129	129	129	120	0	0	0	0	1	0
15. INSTRUMENTATION SYSTEM 23	23	23	5	0	0	0	0	0	0
TOTAL M.A.C. SPECIAL 578	564	559	328	2	3	10	3	11	10
GOVERNMENT FURNISHED EQUIPMENT									
1. GROUND HANDLING 5	4	4	4	1	1	1	0	0	0
2. ALIGNMENT AND MEASUREMENT 1	1	1	1	0	0	0	0	0	0
3. BIOLOGICAL SUPPORT SYSTEM 11	11	11	0	0	0	0	0	0	0
4. OPERATIONAL RECOVERY SYSTEM 9	9	9	0	0	0	0	0	0	0
5. PYROTECHNIC SYSTEM 1	1	1	1	0	0	0	0	0	0
6. REACTION CONTROL SYSTEM 3	3	3	3	0	0	0	0	0	0
TOTAL GFE 30	29	29	9	1	1	1	0	0	0
VENDOR – STANDARD									
1. BIOLOGICAL SYSTEM 2	2	2	2	0	0	0	0	0	0
2. COMMUNICATIONS SYSTEM 3	3	3	2	0	0	0	0	0	0
3. COMPLEX CABLING 1	1	1	1	0	0	0	0	0	0
4. ELECTRICAL SYSTEM 3	3	3	3	0	0	0	0	0	0
5. OPERATIONAL RECOVERY SYSTEM 2	2	2	0	0	0	0	0	0	0
TOTAL VENDOR 11	11	11	8	0	0	0	0	0	0
SUMMARY									
TOTAL GEMINI AGE 609									
TOTAL FERRY AGE 610	604			3			3		
TOTAL FS AGE 614		599			4			11	
TOTAL US AGE 366			345			11			10

F = FERRY
FS = FERRY/SUPPLY
US = UNMANNED SUPPLY

13. OPERATIONS ANALYSIS

Various launch schedules for the station, Ferry and Supply Spacecraft were investigated to determine the minimum number of each type of spacecraft required to accomplish the MORL mission. Estimates of overall reliabilities, for a one year MORL mission, associated with each launch schedule were computed. A cost versus reliability comparison was then made for each schedule to determine the number of spacecraft and launch vehicles, including back-ups, and the associated total recurring cost as a function of overall one year mission reliability. From these data, the least expensive method for obtaining a given mission reliability can be determined, together with the numbers and types of spacecraft and launch vehicles needed.

13.1 Station On-Board Supplies

13.1.1 Station Launch Modes - The amount of supplies that can be launched with the station depends on the manner of launch. Two launch vehicles are considered, Saturn I and Saturn IB, each of which can be launched in two ways: (1) directly into a 250 na. mi. orbit or (2) into an elliptical orbit which is circularized, using on-board propulsion, to a 250 na. mi. orbit.

Table 13.1-1 shows station gross weights for four modes of station launch.

TABLE 13.1-1
LAUNCH VEHICLE PAYLOADS

LAUNCH MODE	WEIGHT – LB.	
	SATURN I	SATURN IB
DIRECT INTO 250 NA.MI. ORBIT	18,100	23,900
100–250 NA.MI. ELLIPTICAL ORBIT	21,650	31,700

13.1.2 Station Supplies and Equipment - Minimum supplies assumed to be on-board the station at launch are:

13.1.2 (Continued)

A.	Scientific and experimental gear	1,000 lbs.
B.	Medical equipment	100 lbs.
C.	Fifteen (15) days expendables (unmanned)	340 lbs.
D.	Spares and repair equipment	800 lbs.
E.	Artificial "g" propellant, tankage and system	654 lbs.
F.	Repressurization and contingency	198 lbs.
G.	Thirty (30) days expendables (manned)	1,350 lbs.
	Total Supplies and Equipment	4,442 lbs.

13.1.3 Allowances for Growth - In addition to the minimum amounts of supplies and equipment, a margin for growth is reserved in the station. When the MORL station is launched by either a Saturn I or Saturn IB directly into a 250 nautical mile orbit, the margin for growth is taken to be 6.5% of the launch vehicle gross payload. This value was selected because it represents the available margin when the station, with an assumed empty weight of 12,500 pounds and with the supplies listed in Section 13.1.2 is launched direct into a 250 na. mi. orbit by the Saturn I. However, when the MORL is launched by either a Saturn I or Saturn IB launch vehicle into an elliptical catch-up orbit with 100 na. mi. perigee and then circularized into a 250 nautical mile orbit, 20% of the gross payload is considered to be reserved as a margin for growth since it represents a more reasonable value. Table 13.1-2 gives a summary of the percentage and quantity of the gross weight that is reserved for future station growth.

TABLE 13.1-2
MARGIN FOR GROWTH ALLOWANCE SUMMARY

LAUNCH MODE	SATURN I		SATURN IB	
	PERCENT OF LAUNCH PAYLOAD	PAYLOAD LB.	PERCENT OF LAUNCH PAYLOAD	PAYLOAD LB.
DIRECT INTO 250 N.M. ORBIT	6.5%	1,177	6.5%	1,654
100–250 NA.MI. ELLIPTICAL ORBIT	20.0%	4,330	20.0%	6,340

13.1.4 Additional Supplies - For the Saturn I station launch, either direct or elliptical, with the margin for growth allowances designated, there is no payload remaining that can be allocated to additional supplies.

For the Saturn IB station launch there is additional payload available for supplies that can be launched with the station. When launch is directly into a 250 nautical mile orbit, there are about 110 days of additional supplies. When the elliptical launch technique is used, there are about 180 days of additional supplies. The rate of use of expendable supplies is taken to be 45 pounds per day as derived in Section 4.3

Table 13.1-3 summarizes the total days of on-board supplies for each mode of launch.

TABLE 13.1-3
SUPPLIES ON-BOARD STATION AT LAUNCH

LAUNCH MODE	SATURN I DAYS SUPPLIES* ON-BOARD STATION	SATURN IB DAYS SUPPLIES* ON-BOARD STATION
DIRECT INTO 250 NA.MI. ORBIT	30	140
100–250 NA.MI. ELLIPTICAL ORBIT	30	210

*ONE DAY'S SUPPLY = 45 LB./ 4 MEN/DAY

13.2 Launch Schedules

13.2.1 Ferry Spacecraft - The Ferry Spacecraft schedule of launches, shown in Table 13.2-1, is taken to be the same as the Ferry portion of the typical mission schedule illustrated in Reference 13.2-1.

General guidelines for the Ferry launches are as follows:

A. The station is not under-staffed after the second (T + 30 days) manned launch.

B. The station is not to be over-staffed, (more than four men), at any time.

13.2.1 (Continued)

C. There are a total of five Ferry launches.

D. A re-entry spacecraft pilot is not to have station time exceeding that for a previously re-entered and examined astronaut.

E. One astronaut remains on-board for one year.

F. Emergency re-entry capability is available for all men at all times.

G. For those launches made on 60, 150, and 240 days, a zero-time pilot will remain on-board the Ferry, returning with the descending astronaut.

H. The MORL mission duration is 360 days.

I. The Ferry is launched by the Gemini Launch Vehicle (GLV).

TABLE 13.2-1
TYPICAL CREW ROTATION SCHEDULE

LAUNCH	TIME (DAYS)	RENDEZVOUS		RETURN		CREWMEN IN LAB	CREW TIME IN LAB (DAYS)
		SPACECRAFT NO.	PILOTS	SPACECRAFT NO.	PILOTS		
LABORATORY SPACECRAFT	-15	–	–	–		–	
1ST FERRY SPACECRAFT	0	1	A & B	–	–	A & B	A=360 B= 60
2ND FERRY SPACECRAFT	+30	2	C & D	–	–	A, B, C & D	C=120 D=210
3RD FERRY SPACECRAFT	+60	3	E & F	1	B & F	A, C, D & E	E=300 F= 0
4TH FERRY SPACECRAFT	+150	4	G & H	2	C & H	A, D, E & G	G=210 H= 0
5TH FERRY SPACECRAFT	+240	5	I & J	3	D & J	A, E, G & I	I=120 J= 0
END OF MISSION	+360	–	–	4 & 5	A, E, G & I	–	

13.2.2 <u>Ferry/Supply Spacecraft</u> - Ferry/Supply Spacecraft can be launched by either a Saturn I or Saturn IB launch vehicle. When a Ferry/Supply Spacecraft is used, it replaces one of the Ferry Spacecraft launches. It is assumed that Ferry/Supply Spacecraft are launched into an elliptical orbit and circularized to

13.2.2 (Continued)

a 250 nautical mile orbit. The amount of supplies on-board the station at launch is an important factor in determining the number of Ferry/Supply Spacecraft required. For example, when the station is launched by a Saturn I into a 250 nautical mile orbit, two Ferry/Supply vehicles launched by Saturn I are needed. If the station were launched by a Saturn IB, then only one Ferry/Supply Spacecraft launched by Saturn I is necessary. In deriving the number of supply launches, the rate of use of supplies of 45 pounds per day per four men (Section 4.3) is used. This rate does not include additional scientific equipment. A requirement for a finite rate of supply of additional scientific equipment could increase the number of launches needed. The total supply payloads used in this analysis are shown in Table 13.2-2. The effects of station launch modes on the number of Ferry/Supply Spacecraft required are indicated in Table 13.2-3.

TABLE 13.2-2
SUPPLY PAYLOADS

CONFIGURATION	SUPPLY PAYLOAD – LB.
FERRY SPACE VEHICLE	
GLV & FERRY	NONE
FERRY/SUPPLY SPACE VEHICLE	
SATURN I & FERRY/ SUPPLY	11,000
SATURN IB & FERRY/SUPPLY	19,400
UNMANNED SUPPLY SPACE VEHICLE	
GLV & STRIPPED GEMINI	4,680
GLV & SPECIFIC DESIGN	5,160

13.2.3 Unmanned Supply Spacecraft - The two unmanned supply spacecraft considered are the Stripped Gemini and the Specific Design, each of which is launched by a GLV.

Again the number of supply spacecraft required to perform the one year MORL mission is directly dependent on the quantity of supplies on-board the station at launch. Unmanned Supply payloads used in this analysis are shown in Table 13.2-2.

13.2.3 (Continued)

TABLE 13.2-3
NUMBERS OF VEHICLES FOR VARIOUS STATION LAUNCH MODES.

CONFIGURATIONS	NUMBER OF VEHICLES IF STATION IS LAUNCHED BY: SATURN I DIRECT	SATURN I ELLIPTICAL	SATURN IB DIRECT	SATURN IB ELLIPTICAL
FERRY & FERRY/SUPPLY SPACE VEHICLE				
SATURN I & FERRY	2	2	1	1
GLV & FERRY	3	3	4	4
SATURN IB & FERRY	1	1	1	1
GLV & FERRY	4	4	4	4
FERRY & UNMANNED SUPPLY SPACE VEHICLE				
GLV & STRIPPED GEMINI	3	3	2	2
GLV & FERRY	5	5	5	5
GLV & SPECIFIC DESIGN	3	3	2	2
GLV & FERRY	5	5	5	5

13.2.4 Total System - The number of space vehicles needed for supply and crew rotation are shown in Table 13.2-3 for various station launch modes and types of supply spacecraft. There are six associated schedules which are shown in Table 13.2-4. The schedules are arranged so that supplies are launched approximately thirty days before the supplies in the station would have been exhausted.

TABLE 13.2-4
LAUNCH SCHEDULES

FERRY/SUPPLY SPACECRAFT

SCHEDULE A SATURN I STATION LAUNCH DIRECT OR ELLIPTICAL (F/S) – SATURN I (F) – GLV	SCHEDULE B SATURN I STATION LAUNCH DIRECT OR ELLIPTICAL (F/S) – SATURN IB (F) – GLV	SCHEDULE C SATURN IB STATION LAUNCH DIRECT OR ELLIPTICAL (F/S) – SATURN I OR SATURN IB (F) – GLV
F 0	F 0	F 0
F/S 30	F/S 30	F/S 30*
F 60	F 60	F 60
F/S 150	F 150	F 150
F 240	F 240	F 240

UNMANNED SUPPLY SPACECRAFT

SCHEDULE D SATURN I STATION LAUNCH DIRECT OR ELLIPTICAL (F) – GLV (S) – GLV	SCHEDULE E SATURN IB STATION LAUNCH DIRECT (ONLY) (F) – GLV (S) – GLV	SCHEDULE F SATURN IB STATION LAUNCH ELLIPTICAL (ONLY) (F) – GLV (S) – GLV
F 0	F 0	F 0
S 15	F 30	F 30
F 30	F 60	F 60
F 60	S 110	F 150
S 110	F 150	S 200
F 150	S 190	F 240
S 210	F 240	S 270
F 240		

ABBREVIATIONS:
(F) = FERRY SPACECRAFT
(F/S) = FERRY/SUPPLY SPACECRAFT
(S) = STRIPPED GEMINI OR SPECIFIC MODULE
*COULD BE LAUNCHED AT DAY + 60

13.3 Systems Comparison - A comparison of the various spacecraft - launch vehicle systems described in Table 13.2-4 was made on the basis of the minimum total recurring cost required to obtain a 0.90 overall mission reliability.

13.3.1 Configurations Cost - The configuration costs used are presented in Table 13.3-1. These are launched costs and include recurring costs for vehicles, spares, launch service costs, transportation and propellants.

TABLE 13.3-1

RECURRING COSTS OF SPACE VEHICLES

SPACE VEHICLE	LAUNCH VEHICLE COST	SPACECRAFT COST	SPACE VEHICLE COST
FERRY SPACE VEHICLE			
GLV & FERRY	\$11.0M	\$12.6M	\$23.6M
FERRY/SUPPLY SPACE VEHICLE (1)			
SATURN I & FERRY/SUPPLY	\$25.0M	\$17.1M	\$42.1M
SATURN IB & FERRY/SUPPLY	\$30.0M	\$17.1M	\$47.1M
UNMANNED SUPPLY SPACE VEHICLE			
GLV & STRIPPED GEMINI	\$11.0M	\$9.2M	\$20.2M
GLV & SPECIFIC DESIGN	\$11.0M	\$9.2M	\$20.2M

(1) INCLUDES \$3.5 MILLIONS FOR FERRY/SUPPLY MODULE
M = MILLIONS OF DOLLARS

13.3.2 Reliability - The model for computing the total system reliability is:

$$R_{Total} = R_{MORL} \times R_{Ferry} \times R_{Ferry/Supply\ or\ Unmanned\ Supply}$$

where: R_{Total} = Total system probability of successfully performing the one year MORL mission.

R_{Ferry} = Total reliability of all Ferry spacecraft and launch vehicles used.

$R_{Ferry/Supply}$ = Total reliability of all Ferry/Supply spacecraft and launch vehicles used.

$R_{Unmanned\ Supply}$ = Total reliability of all unmanned supply spacecraft and launch vehicles used.

13.3.2 (Continued)

R_{MORL} = Probability that the MORL will operate for one year after being successfully put into orbit and activated, assumed to be 0.95.

The reliabilities for the spacecraft are based on Gemini data. The reliability of the launch vehicles are preliminary estimates based on data discussed informally with NASA (Table 13.3-2).

TABLE 13.3-2

LAUNCH VEHICLE AND SPACECRAFT RELIABILITIES

LAUNCH VEHICLE	RELI-ABILITY	SPACECRAFT	RELI-ABILITY	SPACE VEHICLE	RELI-ABILITY
SATURN I	.8400	F/S	.9433	GLV & FERRY	.8607
SATURN IB	.8700	FERRY	.9563	SATURN I & F/S	.7924
GLV	.9000	STRIPPED GEMINI	.9388	SATURN IB & F/S	.8207
		SPECIFIC DESIGN	.9388	GLV & STRIPPED GEMINI	.8449
				GLV & SPECIFIC DESIGN	.8449

The minimum number of launch vehicles and spacecraft needed to perform the mission along with the total reliability are shown in Table 13.3-3 for the schedules shown in Table 13.2-4.

TABLE 13.3-3

SYSTEM RELIABILITY AND COST
NO BACK-UP

SCHEDULE	MINIMUM NUMBER OF SPACE VEHICLES REQUIRED	SYSTEM RELIABILITY	SYSTEM COST
A	3 (F) + 2 (F/S)SATURN I	.3803	$155.0M
B	4 (F) + 1 (F/S) SATURN IB	.4279	$141.5M
C	4 (F) + 1 (F/S) SATURN I/SATURN IB	.4132/.4279	$136.5/141.5M
D	5 (F) + 3 (US)	.2708	$178.6M
E	5 (F) + 2 (US)	.3203	$158.4M
F	5 (F) + 2 (US)	.3203	$158.4M

F = FERRY; F/S = SATURN I OR IB FERRY/SUPPLY.
US = UNMANNED SUPPLY, STRIPPED GEMINI OR SPECIFIC DESIGN.
M = MILLIONS OF DOLLARS

13.3.3 <u>Back-Up Space Vehicles</u> - In order to achieve the 0.90 mission reliability, back-up space vehicles are required for each schedule. To facilitate computation of the total system reliabilities for various combinations of back-up spacecraft and launch vehicles satisfying the minimum schedule requirements (Table 13.3-3), the schedules are divided into two categories: (1) those using Ferry/Supply Spacecraft, and (2) those using Unmanned Supply Spacecraft. In category (1), complete back-up space vehicles, i.e., combined launch vehicle and spacecraft, are assumed. In category (2), back-ups for the launch vehicles and spacecraft are considered separately since a GLV is used to launch Ferry and Unmanned Supply Spacecraft and, hence, can back-up either type of launch.

System reliabilities and associated costs of various combinations of vehicles are shown in Tables 13.3-4 through 13.3-8 for the six schedules. The least expensive combination for a reliability of 0.90 is indicated.

TABLE 13.3–4

BACK-UP RELIABILITY – SCHEDULE 'A'

NUMBER OF BACK-UP FERRY/SUPPLY SPACE VEHICLES	NUMBER OF BACK-UP FERRY SPACE VEHICLES	TOTAL SYSTEM RELIABILITY	BACK-UP COST
0	0	.3803	—
1	0	.5383	$42.1M
0	1	.5393	23.6M
0	2	.5836	47.2M
2	0	.5874	84.2M
0	3	.5938	70.8M
3	0	.6009	126.3M
1	1	.7632	65.7M
1	2	.8258	89.3M
2	1	.8330	107.8M
1	3	.8403	112.9M
3	1	.8521	149.9M
2	2	.9014	131.4M
2	3	.9170	155.0M
3	2	.9220	173.5M
3	3	.9380	197.1M

M = MILLIONS OF DOLLARS

TABLE 13.3–5

BACK-UP RELIABILITY – SCHEDULE 'B & C'★

NUMBER OF BACK-UP FERRY/SUPPLY SPACE VEHICLES	NUMBER OF BACK-UP FERRY SPACE VEHICLES	TOTAL SYSTEM RELIABILITY	BACK-UP COST
0	0	.4279	——
1	0	.5046	$47.1M
2	0	.5183	94.2M
0	1	.6663	23.6M
0	2	.7494	47.2M
0	3	.7724	70.8M
0	4	.7781	94.4M
1	1	.7858	70.7M
2	1	.8071	117.8M
1	2	.8837	94.3M
2	2	.9076	141.4M
1	3	.9110	117.9M
1	4	.9177	141.5M
2	3	.9357	165.0M
2	4	.9425	188.6M

M = MILLIONS OF DOLLARS

★ SATURN IB FERRY/SUPPLY USED

TABLE 13.3–6

BACK-UP RELIABILITY – SCHEDULE 'C'★

NUMBER OF BACK-UP FERRY/SUPPLY SPACE VEHICLES	NUMBER OF BACK-UP FERRY SPACE VEHICLES	TOTAL SYSTEM RELIABILITY	BACK-UP COST
0	0	.4132	——
1	0	.4988	$42.1M
2	0	.5167	84.2M
0	1	.6433	23.6M
0	2	.7235	47.2M
0	3	.7458	70.8M
0	4	.7513	94.4M
1	1	.7769	65.7M
2	1	.8047	107.8M
1	2	.8737	89.3M
2	2	.9049	131.4M
1	3	.9063	112.9M
1	4	.9073	136.5M
2	3	.9328	155.0M
2	4	.9396	178.6M

M = MILLIONS OF DOLLARS

★ SATURN I FERRY/SUPPLY USED

TABLE 13.3–7

BACK-UP RELIABILITY – SCHEDULE 'D'

NUMBER OF BACK-UP GLV	NUMBER OF BACK-UP UNMANNED SUPPLY SPACECRAFT	NUMBER OF BACK-UP FERRY SPACECRAFT	TOTAL SYSTEM RELIABILITY	BACK-UP COST
0	0	0	.2708	——
1	1	0	.3966	$20.2M
1	0	1	.4592	23.6M
1	1	1	.5852	32.8M
2	2	0	.4358	40.4M
2	0	2	.5469	47.2M
2	2	1	.7120	53.0M
2	1	2	.7909	56.4M
2	2	2	.8185	65.6M
3	3	0	.4458	60.6M
3	0	3	.5525	70.8M
3	2	1	.7397	64.0M
3	3	1	.7499	73.2M
3	1	2	.7882	67.4M
3	1	3	.8028	80.0M
3	2	2	.8549	76.6M
3	3	2	.8650	85.8M
3	2	3	.8696	89.2M
3	3	3	.8959	98.4M
4	3	3	.9077	109.4M
4	3	4	.9079	120.0M

M = MILLIONS OF DOLLARS

TABLE 13.3–8

BACK-UP RELIABILITY – SCHEDULE 'E & F'

NUMBER OF BACK-UP GLV	NUMBER OF BACK-UP UNMANNED SUPPLY SPACECRAFT	NUMBER OF BACK-UP FERRY SPACECRAFT	TOTAL SYSTEM RELIABILITY	BACK-UP COST
0	0	0	.3203	——
1	1	0	.4197	$20.2M
1	0	1	.5435	23.6M
1	1	1	.6429	32.8M
2	2	0	.4429	40.4M
2	0	2	.6367	47.2M
2	2	1	.7352	53.0M
2	1	2	.8052	56.4M
2	2	2	.8284	65.6M
3	3	0	.6416	60.6M
3	0	3	.6670	70.8M
3	2	1	.7514	64.0M
3	3	1	.7561	73.2M
3	1	2	.8476	67.4M
3	1	3	.8778	80.0M
3	2	2	.8868	76.6M
3	3	2	.8916	85.8M
3	2	3	.9171	89.2M
3	3	3	.9219	98.4M

M = MILLIONS OF DOLLARS

13.3.4 Systems Evaluation - The minimum system costs for various levels of reliability are shown in Table 13.3-9 and Figure 13.3-1. For each value of reliability, the combination of space vehicles which results in the least cost is shown.

A distinction is made between Saturn IB and Saturn I station launches. For Saturn IB station launches, supply by either Saturn I, Saturn IB or GLV launched spacecraft is considered. For Saturn I station launches, the same possibilities are shown for supply launches, together with one which does not consider Saturn IB supply launches. This latter case is considered, since a Saturn I station launch implies unavailability of Saturn IB launch vehicles.

The reliability and cost data for combinations having a reliability of about 0.90, assumed to be the MORL mission goal, are summarized in Table 13.3-10.

TABLE 13.3-9

MINIMUM SYSTEM RECURRING COST FOR MAXIMUM SYSTEM RELIABILITY

SYSTEM COST (MILLION $)	RELIABILITY	SCHEDULE REFERENCE	NORMAL* LAUNCH	NUMBER OF VEHICLES REQUIRED**			
				FERRY	UNMANNED SUPPLY	GLV	SATURN (I OR IB)
SATURN IB STATION LAUNCH (SATURN I, IB OR GLV SUPPLY LAUNCHES)							
137	.41	C	4F-1F/S	5		4	1 (I)
142	.43	C	4F-1F/S	5		4	1 (IB)
160	.64	C	4F-1F/S	6		5	1 (I)
165	.67	C	4F-1F/S	6		5	1 (IB)
184	.72	C	4F-1F/S	7		6	1 (I)
189	.75	C	4F-1F/S	7		6	1 (IB)
202	.78	C	4F-1F/S	7		5	2 (I)
212	.79	C	4F-1F/S	7		5	2 (IB)
215	.81	E OR F	5F-2US	7	3	9	
224	.83	E OR F	5F-2US	7	4	9	
226	.87	C	4F-1F/S	8		6	2 (I)
235	.89	E OR F	5F-2US	7	4	10	
248	.92	E OR F	5F-2US	8	4	10	
307	.94	C	4F-1F/S	10		7	3 (IB)
SATURN I STATION LAUNCH (SATURN I, IB OR GLV SUPPLY LAUNCHES)							
142	.43	B	4F-1F/S	5		4	1 (IB)
165	.67	B	4F-1F/S	6		5	1 (IB)
189	.75	B	4F-1F/S	7		6	1 (IB)
212	.79	B	4F-1F/S	7		5	2 (IB)
236	.88	B	4F-1F/S	8		6	2 (IB)
259	.91	B	4F-1F/S	9		7	2 (IB)
283	.92	B	4F-1F/S	10		8	2 (IB)
307	.94	B	4F-1F/S	10		7	3 (IB)
SATURN I STATION LAUNCH (SATURN I OR GLV SUPPLY LAUNCHES)							
155	.38	A	3F-2F/S	5		3	2 (I)
179	.54	A	3F-2F/S	6		4	2 (I)
202	.58	A	3F-2F/S	7		5	2 (I)
211	.59	D	5F-3US	6	4	9	
221	.76	A	3F-2F/S	7		4	3 (I)
235	.79	D	5F-3US	7	4	10	
244	.83	A	3F-2F/S	8		5	3 (I)
255	.85	D	5F-3US	7	5	11	
264	.87	D	5F-3US	7	6	11	
277	.90	D	5F-3US	8	6	11	
288	.91	D	5F-3US	8	6	12	
310	.92	A	3F-2F/S	10		6	4 (I)
352	.94	A	3F-2F/S	11		6	5 (I)

* F = FERRY, F/S = SATURN I OR IB FERRY/SUPPLY;
US = UNMANNED SUPPLY, STRIPPED GEMINI OR SPECIFIC DESIGN
** DOES NOT INCLUDE STATION LAUNCH VEHICLE

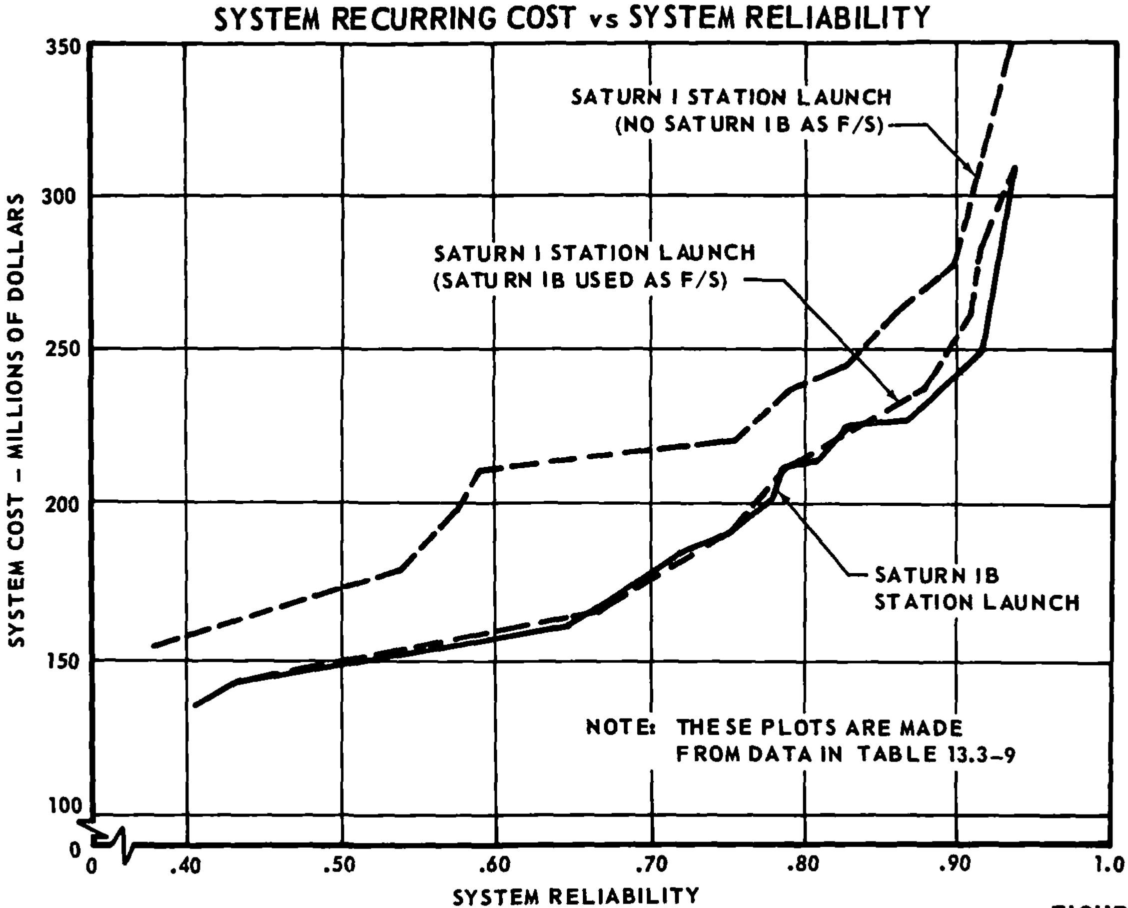


FIGURE 13.3-1

TABLE 13.3-10

SYSTEM COST TO OBTAIN .90 SYSTEM RELIABILITY

SCHEDULE	STATION LAUNCH BY:	LAUNCH (1) VEHICLES REQUIRED		SPACECRAFT REQUIRED		PADS(2) REQUIRED WITH BACK-UPS		SYSTEM COST	SYSTEM RELIABILITY
		SATURN (I OR IB)	GLV	FERRY(3)	UNMANNED(4) SUPPLY	SATURN	GLV		
A	SATURN I	4 (I)	5	9	N/A	2	2	$286.4 M	.9014
B	SATURN I	2 (IB)	7	9	N/A	2	2	$259.4 M	.9110
C	SATURN IB	2 (IB)	7	9	N/A	2	2	$259.4 M	.9110
C	SATURN IB	2 (I)	7	9	N/A	2	2	$249.4 M	.9063
D	SATURN I	N/A	11	8	6	N/A	4	$277.0 M	.8959
E	SATURN IB	N/A	10	8	4	N/A	3	$247.6 M	.9171
F	SATURN IB	N/A	10	8	4	N/A	3	$247.6 M	.9171

M = MILLIONS OF DOLLARS
(1) DOES NOT INCLUDE MORL OR MORL LAUNCH VEHICLE.
(2) SIXTY DAY PAD TURN-AROUND TIME.
(3) A FERRY WITH A SATURN LAUNCH VEHICLE INCLUDES THE SUPPLY MODULE.
(4) SUPPLY SPACECRAFT MEANS EITHER SPECIFIC DESIGN OR STRIPPED GEMINI.

13.3.5 Launch Pads and Turn-Around Times - The number of launch pads required for each of the six schedules with the back-ups shown in Table 13.3-10 were determined based on the following assumptions:

A. Two pads are always available for any launch in the nominal schedules.

B. Pad turn-around times are 30 days (optimistic) and 60 day (normal).

C. The time required to convert a back-up GLV/Ferry Space Vehicle to a GLV/Unmanned Supply Space Vehicle is 6 to 10 days.

D. The time required to convert a back-up GLV/Unmanned Supply Space Vehicle to a GLV/Ferry Space Vehicle is 8 to 14 days.

E. A failure of the first launch on day zero will lead to a rescheduling.

F. The following pads are available at AMR.

1. GLV, numbers 15, 16, 19, and 20.

2. Saturn I or IB, numbers 34, 37A and 37B.

Application of these assumptions is presented in Table 13.3-11 for Schedule 'E', using a 60-day pad turn-around time. The pads required for the six schedules with back-ups for 0.90 reliability are given in Table 13.3-12.

TABLE 13.3–11

LAUNCH PAD USE ILLUSTRATION

ACTIVITY			
TIME (DAYS)	PAD 1	PAD 2	PAD 3
0	P(L)	B–U	—
30	R	P(L)	B–U
60	B–U	R	P(L)
110	P(L)	B–U	R
150	R	P(L)	B–U
190	B–U	R	P(L)
240	P(L)	B–U	—

P(L) = PRIMARY LAUNCH VEHICLE, B–U = BACK-UP; R = BEING REFURBISHED

TABLE 13.3–12

LAUNCH PADS REQUIRED

SCHEDULE	TYPE PADS USED	PAD TURN-AROUND TIMES	
		30 DAYS	60 DAYS
A	SATURN/GLV	2/2	2/2
B	SATURN/GLV	2/2	2/2
C	SATURN/GLV	2/2	2/2
D	GLV	3	4
E	GLV	2	3
F	GLV	2	3

14. RECOMMENDATIONS FOR ADDITIONAL ENGINEERING, RESEARCH AND DEVELOPMENT

The engineering tasks recommended for the continued development of Ferry, Ferry/Supply and Unmanned Supply Spacecraft for the MORL mission are outlined in this section.

These recommendations are based on an early selection of:

A. A single configuration, each, for the Ferry and Ferry/Supply Spacecraft.

B. A single operational orbit altitude.

C. Specific MORL system parameters for analysis, as opposed to general parametric studies.

The major tasks to be accomplished during future work are divided into ten categories. More-or-less routine tasks, such as preparation of design layouts, stress analysis and normal technical analyses are not specifically discussed. No attempt was made to categorize items by development "phase", since phases and their scope require definition by NASA.

14.1 Design - Additional design effort should include investigation of:

A. Modifications to Gemini to accommodate the larger Apollo pressure suit.

B. One-man operation of Ferry Spacecraft to provide cargo space, at launch and re-entry, at the other crew station.

C. Further investigation of spacecraft installations for stowage of experimental material including biomedical samples and photographs in returning Ferry Spacecraft.

14.2 Technical Analyses - Technical analysis is needed in at least three major areas, as follows:

A. Investigation of methods of further reducing the velocity increment needed for rendezvous through utilization of (1) yaw steering of the launch vehicle, (2) up-dated tracking accuracy information in order to re-run in-orbit correction dispersions, and (3) more advanced methods of rendezvous, such as injection into a

14.2 (Continued)

low circular orbit followed by a three-pulse orbital transfer.

B. Further analysis of Saturn launch vehicle escape trajectories, both off the pad and at maximum q, for the escape tower and adapter mounted rocket configurations and selection of one configuration for further development. If additional data can be obtained on Saturn fireball characteristics, further analysis will be made of the heating effects.

C. Continued analysis of the self-inject propulsion system for Ferry and Unmanned Supply Spacecraft, including investigation of abort modes due to self-inject system failure, reliability and range safety.

D. Radiation and meteoroid penetration analysis using updated pressure suit data.

14.3 Standby Investigations - Further investigation of the standby period of the Ferry mission is needed. These would take the form of:

A. Further analysis and definition of the thermal and radiation environments throughout the spacecraft.

B. Additional investigation of equipments by component, and specification of the environmental limitations.

C. Verification of heat input requirements to equipments and of locations of heaters to control standby temperatures.

D. Analysis of the use of low energy radioisotopes (i.e. Alpha emitters) as heat sources during standby.

E. Study of methods of measuring fluid quantities in tanks under zero gravity.

F. Definition of Ferry equipments amenable to in-space maintenance or replacement, necessary tools and procedures and performance of crewmen.

14.4 Guidance and Control Simulation - The study of guidance and control problems would be aided by the following simulations:

A. The terminal phase of rendezvous, to evaluate the effects of loss of a thruster on procedures and system control characteristics.

B. Docking, to evaluate the handling characteristics of the Ferry/Supply Spacecraft.

C. Docking, to evaluate the remote handling characteristics of the Unmanned Supply Spacecraft.

14.5 MORL Interface Coordination - Close coordination with NASA and, at NASA discretion, with the MORL contractor is needed for the determination of:

A. The docking cone installation.

B. The airlock location and ECS extension line provisions.

C. Cargo transfer provisions.

D. Unmanned Supply rendezvous and docking control.

E. Unmanned Supply separation control.

F. Initial manning requirements for MORL.

G. MORL operations affecting ferry design (e.g., hangars and the relocation of a Ferry Spacecraft to another docking port during the standby period).

H. Umbilical requirements.

I. Electronics interface trade-offs (e.g., the use of directional antennas on MORL for operation of a telemetry link between Ferry and MORL, and the effect of such antennas on the MORL attitude and the maneuver envelope of the Ferry.)

14.6 Crew Mobility Studies - Crew transfer and mobility studies should be expanded to include investigation of:

A. General mobility and transfer capability in up-to-date pressure suits of various types (e.g., Gemini, Apollo, Dyna Soar).

B. The ability of crewmen to perform maintenance in pressurized suits.

14.6 (Continued)

C. The increase in crew mobility for short-time operation at reduced suit pressures, including estimation of time limitations.

14.7 Ground Systems Studies - Ground systems studies should include:

A. Further analysis of Aerospace Ground Equipment (AGE) needs for each spacecraft.

B. Investigation of modified checkout procedures which might result in decreasing pad turn-around time.

14.8 Mock-Ups - The following mock-ups would aid in future studies:

A. Mock-ups of cargo packages and supply spacecraft pressurized cargo compartments to evaluate methods of cargo transfer and accurately determine necessary volume. Exploration of the limitations of cargo handling imposed by the small diameter "nose-type" docking cone is included.

B. More detailed (harder) modifications to the Gemini mock-up to include actual Ferry hatch mechanisms and operable mating parts on MORL for use in further crew mobility investigations (Section 14.6) and to aid in definition of the interface.

14.9 Miscellaneous Studies - In addition to the preceding other investigations would include:

A. Analysis of the use of the Ferry Spacecraft for space rescue operations. One method, for example, is the use of an Unmanned Ferry in a manner similar to the use of the Unmanned Supply Spacecraft.

B. Up-dated operations analysis of cost and logistics schedules using launch vehicle reliabilities and unit costs coordinated with NASA.

14.10 Tests - The recommended test program is described in Section 15.

15. TEST PROGRAM, DEVELOPMENT SCHEDULES AND COSTS

A test program will be needed to support the design effort for Ferry and Supply Spacecraft. Much of the ground test program is directed toward the unique aspects of standby in space. A flight test program is recommended where previously untried design features or mission requirements are involved. Acceptance tests similar to those used in the Gemini program are recommended.

A schedule outlining the planned development effort is included. Major activities and milestones are shown as well as the estimated dates of these items.

Estimated program costs are presented for various logistics schedules. Both recurring and non recurring costs are included.

15.1 Test Program - Testing is required in both the Preliminary Design phase (Phase II) and the Development and Delivery phase (Phase III).

15.1.1 Phase II Tests - The Phase II Test program includes only design information tests needed to support design decisions and is based primarily on standby considerations. The schedule for this phase of the Ferry Spacecraft program is shown in Figure 15.1-1. The test items include those whose results, if adverse, could lead to major redesign.

STANDBY CAPABILITY TESTS SCHEDULE
PRELIMINARY DESIGN (PHASE II)

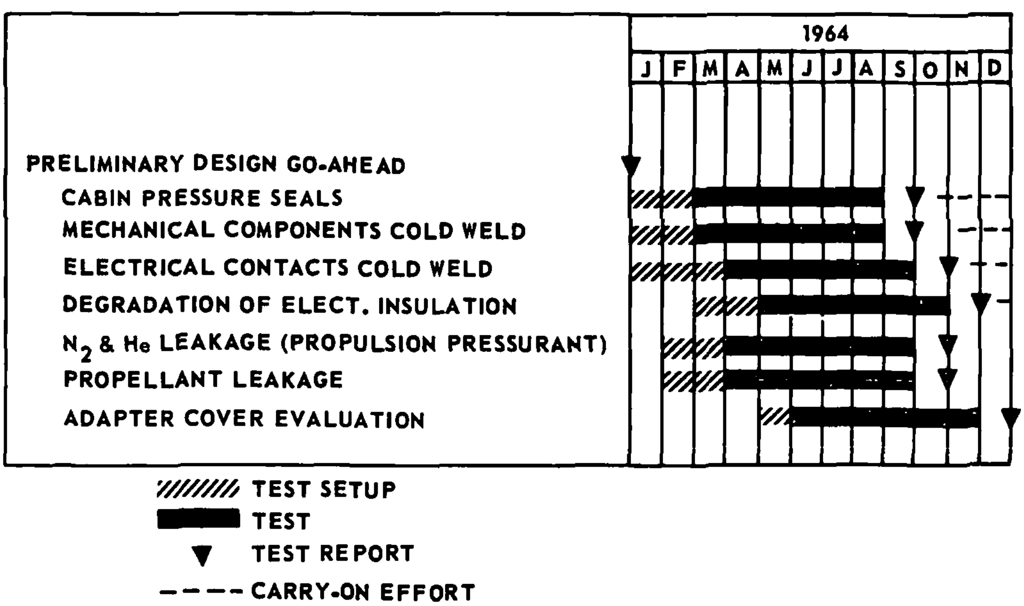


FIGURE 15.1-1

15.1.1 (Continued)

Cabin Pressure Seals - The seals are silicone or silicone/fabric laminates. Compression set of the silicone which would result in seal stress relaxation and increased cabin leakage, is a possible problem. There is also the possibility that the silicone seals (for the hatch, rotary switches and similar items) may bond to the mating seal surface and tear when motion is required. A seal development effort or a change in orbital storage procedure could arise from the results of these tests.

A demonstration test is scheduled early in the program. The test involves using pressurized containers with Gemini sealing techniques exposed to a thermal/vacuum environment and monitoring leakage rates.

Cold Welding - The cold welding test requirements can be divided into two parts, metal/lubricant/metal and metal/metal contacts.

A. Metal/Lubricant/Metal Contact - These tests are conducted on an apparatus which duplicates the Timken straight line contact dry film lubricant test in a thermal/vacuum environment. Since the test apparatus can accommodate only two specimens at a time, it is unrealistic to thermal/vacuum soak and test on a real time basis. Possible solutions to this problem are: (1) development of an apparatus which will test many samples simultaneously, (2) development of an accelerated life test model or, (3) the testing of a reasonable number of specimens under identical conditions to choose the best material.

B. Metal/Metal Contact - Testing consists of subjecting a number of typical switches and circuit breakers to a thermal/vacuum environment and conducting a real-time simulated mission test. Circuit breaker, switch loads and actuation signals would be simulated and monitored.

Electrical Insulation - Prolonged exposure of the wiring insulation (irradiated polyolefin) to the space environment may allow sufficient vaporization of the

15.1.1 (Continued)

plastisizer that the insulation becomes brittle causing failure during the vibration of retro-fire and re-entry.

A thermal/vacuum test program will be conducted to determine degradation of elasticity for an estimated worst case.

Propulsion System - Sources and magnitudes of internal and external leakage and permeation of pressurants and propellants need to be determined and the associated effects assessed. The following are typical leakage and permeation unknowns which should be investigated:

A. Effect of long term propellant storage with temperature induced cycling on the Teflon bladders.

B. Pressurant and propellant margin requirements.

C. Permeation rate of propellants through Teflon bladders.

D. Effect of standby temperature cycling on propellant tank pressure.

The test set-up consists of an instrumented functional system mock-up assembled from production configuration hardware. It is set-up so that test instrumentation used to monitor system parameters does not compromise the results.

Testing is conducted in an ambient pressure controlled temperature enclosure. System temperature history, state of activation and propellant usage determined during design analysis are simulated.

Propellant samples are taken following the simulated mission for ignition and performance characteristics tests.

Adapter Cover - The adapter cover, which is made from an aluminum foil/silicone adhesive/fiberglass cloth laminate, must maintain a low space side emissivity during orbital storage to prevent spacecraft heater power requirements from increasing with time. Emissivity stability with ultraviolet and Lyman-alpha exposure in a 10^{-6} torr or less environment should be determined early in the program. Material

15.1.1 (Continued)

temperature would be held at a nominal temperature determined from the initial emissivity; and the total dosage to be applied will be representative of an actual mission.

It is estimated that the temperature of the sun oriented adapter cover will reach 427°F which may degrade the mechanical properties of the silicone adhesive. A test program to determine the degree of degradation should be conducted.

15.1.2 Phase III Tests

Ground Tests - Selected wind tunnel, development and qualification, structural, dynamic, and system and subsystem demonstration tests, determined to be desirable during Phase II, would be conducted. These would be essentially of a routine nature using techniques and test equipments developed during the Mercury and Gemini programs.

Flight Tests - The type and number of development launches to be conducted during the flight test program are shown in Figure 15.1-2.

For nose dock - nose moor or nose dock - side moor Ferry Spacecraft, the minor changes to Gemini are not considered significant enough to require any new development flights. The higher orbital altitude (250 na. mi. vs. 161 na. mi. for Gemini) causes a re-entry velocity increase of only 300 feet per second over that of Gemini. (Reference Section 8.5.2) The small increase in heating and resulting heat shield redesign are considered minor items; consequently, system integrity can be satisfactorily demonstrated by ground tests and by extrapolation of Gemini technology. Similarly, the Ferry system standby and re-start capability can be substantially demonstrated and evaluated in space environment simulation tests on the ground.

For aft docking Ferry Spacecraft, an unmanned launch is scheduled to prove out the heat shield re-entry capability with the hatch incorporated. As an adjunct test on this flight, the Ferry systems would be shut down temporarily and then

DEVELOPMENT FLIGHT TEST PROGRAM

				TESTS						
SPACECRAFT	LAUNCHES	TYPE	LAUNCH VEHICLE	SYSTEM RE-START AFTER SHUTDOWN	RENDEZVOUS AND DOCKING	PROPULSION SYSTEM	SEPARATION ROCKETS	RE-ENTRY AND RECOVERY	ESCAPE SYSTEM	REMOTE RENDEZVOUS AND DOCKING
FERRY NOSE DOCK–NOSE MOOR	0									
FERRY NOSE DOCK–SIDE MOOR	0									
FERRY AFT DOCK–AFT MOOR	2	1 UNMANNED 1 MANNED	GLV GLV	X —	— X	X X	X X	X X		
FERRY/ SUPPLY NOSE DOCK–NOSE MOOR	5	2 UNMANNED* 2 UNMANNED 1 MANNED	ESCAPE ROCKETS LITTLE JOE II SATURN I OR IB	— — —	— — X	— — X	— — X	— — X	X X 	
FERRY/ SUPPLY NOSE DOCK–SIDE MOOR	5	2 UNMANNED* 2 UNMANNED 1 MANNED	ESCAPE ROCKETS LITTLE JOE II SATURN I OR IB	— — —	— — X	— — X	— — X	— — X	X X 	
FERRY/ SUPPLY AFT DOCK–AFT MOOR	6	2 UNMANNED* 2 UNMANNED 1 UNMANNED 1 MANNED	ESCAPE ROCKETS LITTLE JOE II SATURN I OR IB SATURN I OR IB	— — X —	— — — X	— — X X	— — X X	— — X X	X X 	
UNMANNED SUPPLY STRIPPED GEMINI	1		GLV	—	—	X	X	—	—	X
UNMANNED SUPPLY SPECIFIC DESIGN	1		GLV	—	—	X	X	—	—	X

*BOILERPLATE

FIGURE 15.1–2

15.1.2 (Continued)

re-started to simulate a Ferry return after standby. A manned launch would follow, primarily to checkout aft docking and mooring techniques. Either an early developmental MORL or an Agena D type vehicle could be used as the orbiting target.

For any Ferry/Supply Spacecraft, four launches would be made to demonstrate escape system adequacy. Two launches from a fixed tower using the spacecraft escape rockets on a boilerplate spacecraft would simulate an escape from a launch vehicle explosion on the pad. Two maximum aerodynamic pressure (maximum q) escape system test flights are scheduled using Little Joe II launch vehicles.

A manned Saturn launch for the Ferry/Supply Spacecraft is included to prove-out the rendezvous, docking and mooring with the increased vehicle mass and redesigned propulsion system.

Since the tests for the aft dock - aft moor Ferry/Supply Spacecraft include those for the aft dock - aft moor Ferry, Gemini Launch Vehicle flight tests would not be needed for a program which uses both these spacecraft.

For Unmanned Supply Spacecraft, one test launch would suffice to test the new propulsion systems (self-inject, orbit attitude and maneuver system, and orbit maintenance)and the remote rendezvous and docking feature. It is anticipated that the remote rendezvous and docking tests can be controlled from the ground using an unmanned target. If this is shown to be unfeasible, then a manned Ferry Spacecraft would be required in orbit to conduct this test.

Acceptance Tests

A. Pre-installation Acceptance Testing (PIA) - PIA testing conducted to the philosophy and procedures developed in the Gemini program is scheduled.

B. Spacecraft Systems Test (SST) - The purpose of SST's is to assume delivery of a properly assembled and operationally correct production spacecraft to the

15.1.2 (Continued)

launch site. The test procedures, personnel, AGE and facilities used for Gemini would be used with appropriate modifications for the MORL program.

15.2 Development Schedule - The Ferry and Supply Spacecraft development schedule is presented in Figure 15.2-1. The schedule is divided into two parts: Preliminary Design (Phase II), and Development and Delivery (Phase III). Under the Development Delivery phase, schedules of three different groups of hardware are shown. Group A represents the minimum change items, Group B the moderate change items, and Group C the maximum change items. These groups are typified, respectively, by the nose dock - nose moor Ferry, the nose dock - nose moor Ferry with supply adapter, and the aft dock - aft moor Ferry.

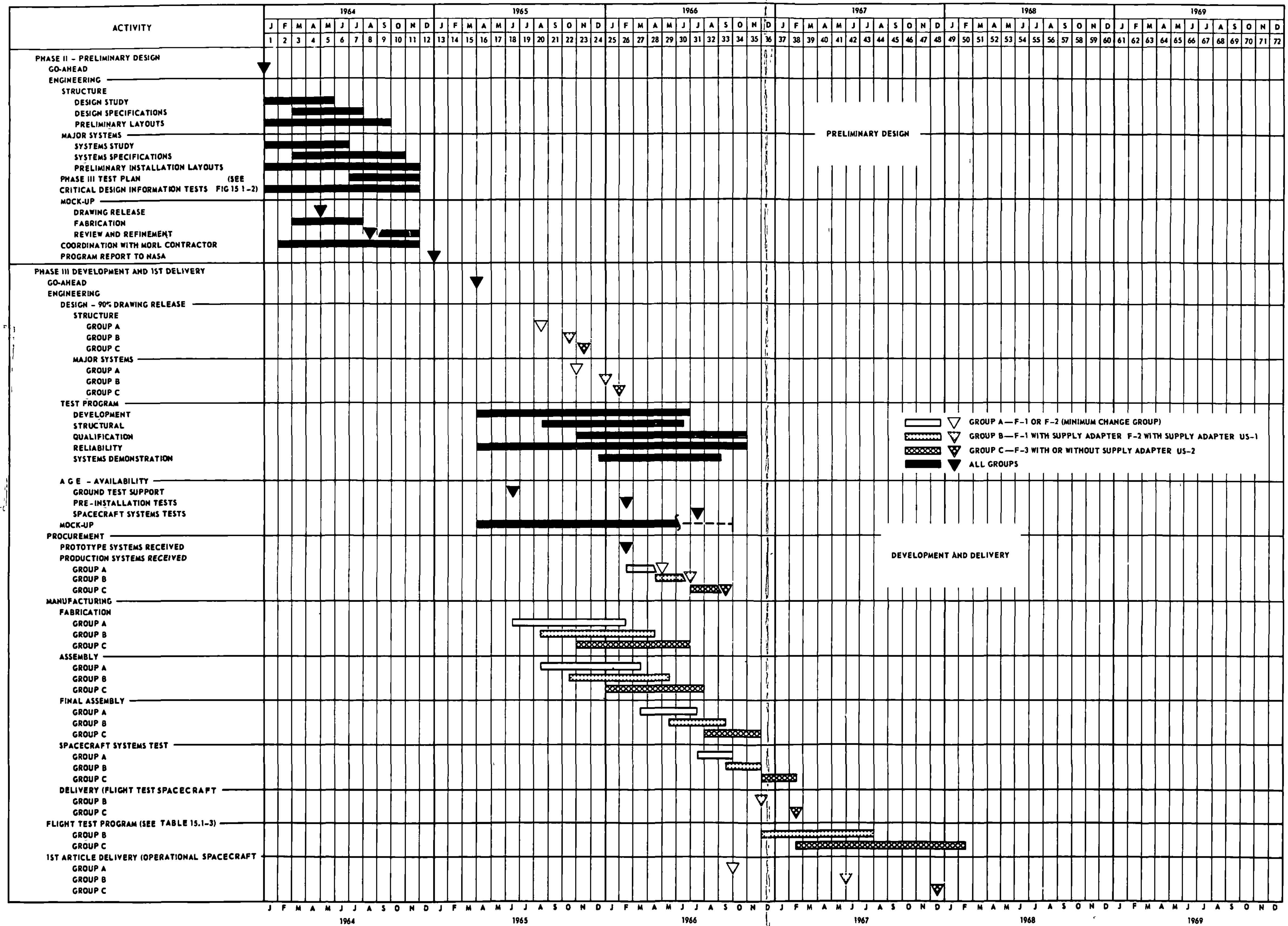
Timing of all programs is based on a Phase II go-ahead in January 1964.

In the manufacturing phase, equal amounts of time are shown for each group. For example, eight months are required for fabrication regardless of which Group (A, B or C) is selected. This time represents the minimum time that the particular operation would require for the minimum change group. For any other group more personnel would be used to accomplish the increased work within the specified time.

The earliest first article delivery date of any group is October 1966 (Group A). This delivery would support a MORL launch in January 1967. For the maximum change spacecraft (Group C) the first article delivery date is December 1967, which would support a MORL launch in March 1968.

15.3 Costs - The estimated program costs are presented in Figure 15.3-1. These costs are based upon 1964 dollars (i.e., no predicted inflationary trends are included). Both recurring and non-recurring costs are included for the various operational schedules established in Section 13. Included are the costs of back-up spacecraft and launch vehicles required to obtain an overall MORL mission reliability of 0.90. Because of the many spacecraft configurations (nose dock, aft dock, etc.) and combinations of the various spacecraft and launch vehicles,

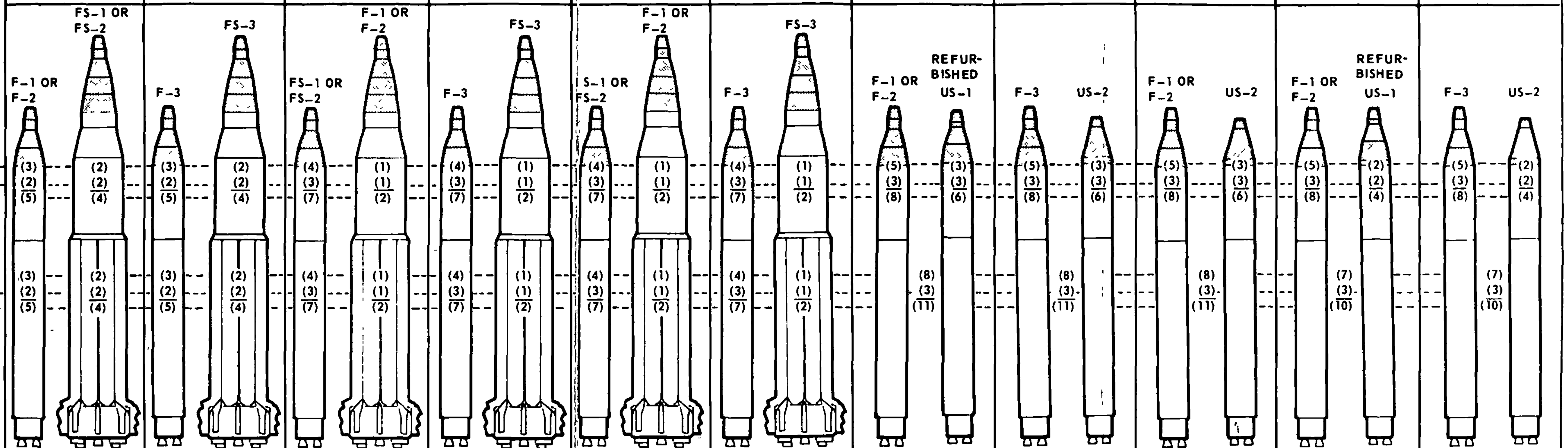
DEVELOPMENT SCHEDULE
FERRY AND SUPPLY SPACECRAFT



NOTE F-1 = FERRY (NOSE DOCK - NOSE MOOR), F-2 = FERRY (NOSE DOCK - SIDE MOOR), F-3 = FERRY (AFT DOCK - AFT MOOR), US-1 = UNMANNED SUPPLY (STRIPPED GEMINI) US-2 = UNMANNED SUPPLY (SPECIFIC DESIGN)

FIGURE 15.2-1

FERRY PROGRAM COST
(MILLIONS OF DOLLARS)



Schedule / Combination	Vehicle	Launch vehicle	Spacecraft scheduled	Spacecraft back-ups required *	Spacecraft total	Launch vehicle scheduled	Launch vehicle back-ups required *	Launch vehicle total
SCHEDULE A – COMBINATION 1	F-1 OR F-2	GLV	(3)	(2)	(5)	(3)	(2)	(5)
	FS-1 OR FS-2	S-I	(2)	(2)	(4)	(2)	(2)	(4)
SCHEDULE A – COMBINATION 2	F-3	GLV	(3)	(2)	(5)	(3)	(2)	(5)
	FS-3	S-I	(2)	(2)	(4)	(2)	(2)	(4)
SCHEDULE B – COMBINATION 1	FS-1 OR FS-2	GLV	(4)	(3)	(7)	(4)	(3)	(7)
	F-1 OR F-2	S-IB	(1)	(1)	(2)	(1)	(1)	(2)
SCHEDULE B – COMBINATION 2	F-3	GLV	(4)	(3)	(7)	(4)	(3)	(7)
	FS-3	S-IB	(1)	(1)	(2)	(1)	(1)	(2)
SCHEDULE C – COMBINATION 1	S-1 OR FS-2	GLV	(4)	(3)	(7)	(4)	(3)	(7)
	F-1 OR F-2	S-I	(1)	(1)	(2)	(1)	(1)	(2)
SCHEDULE C – COMBINATION 2	F-3	GLV	(4)	(3)	(7)	(4)	(3)	(7)
	FS-3	S-I	(1)	(1)	(2)	(1)	(1)	(2)
SCHEDULE D – COMBINATION 1	F-1 OR F-2	GLV	(5)	(3)	(8)	(8)	(3)	(11)
	REFURBISHED US-1	GLV	(3)	(3)	(6)			
SCHEDULE D – COMBINATION 2	F-3	GLV	(5)	(3)	(8)	(8)	(3)	(11)
	US-2	GLV	(3)	(3)	(6)			
SCHEDULE D – COMBINATION 3	F-1 OR F-2	GLV	(5)	(3)	(8)	(8)	(3)	(11)
	US-2	GLV	(3)	(3)	(6)			
SCHEDULE E & F – COMBINATION 1	F-1 OR F-2	GLV	(5)	(3)	(8)	(7)	(3)	(10)
	REFURBISHED US-1	GLV	(2)	(2)	(4)			
SCHEDULE E & F – COMBINATION 2	F-3	GLV	(5)	(3)	(8)	(7)	(3)	(10)
	US-2	GLV	(2)	(2)	(4)			

SCHEDULE A

	COMBINATION 1 (GLV / S-I) R	NR	TOTAL	COMBINATION 2 (GLV / S-I) R	NR	TOTAL
FERRY RE-ENTRY MODULES	(9)62.0	25.4	87.4	(9)65.2	30.0	95.2
FERRY ADAPTERS	(9)14.9	8.7	23.6	(9)18.8	17.1	35.9
SUPPLY MODULES	(4)13.2	32.5	45.7	(4)13.2	32.5	45.7
UNMANNED SUPPLY SPACECRAFT						
FERRY/SUPPLY LAUNCH ESCAPE SYSTEM	(4) 2.8	6.9	9.7	(4) 2.8	6.9	9.7
SPACECRAFT SPARES	18.6		18.6	20.0		20.0
SPACECRAFT LAUNCH SERVICE	21.8		21.8	22.7		22.7
SPACECRAFT PROPELLANT & TRANSPORTATION	1.8		1.8	1.9		1.9
GEMINI LAUNCH VEHICLES	(5) 55.0		55.0	(5) 55.0		55.0
SATURN LAUNCH VEHICLES	(4)100.0		100.0	(4)100.0		100.0
A.G.E. (FERRY & SUPPLY)		54.6	54.6		56.7	56.7
LAUNCH PAD CONVERSION		0.8	0.8		0.8	0.8
TRAINERS & SUPPORT		0.8	0.8		1.1	1.1
DEVELOPMENT FLIGHT TEST		89.0	89.0		133.4	133.4
TOTAL PROGRAM COST →			508.8			578.1

SCHEDULE B

	COMBINATION 1 (GLV / S-IB) R	NR	TOTAL	COMBINATION 2 (GLV / S-IB) R	NR	TOTAL
FERRY RE-ENTRY MODULES	(9)62.0	25.4	87.4	(9)65.2	30.0	95.2
FERRY ADAPTERS	(9)14.9	8.7	23.6	(9)18.8	17.1	35.9
SUPPLY MODULES	(2) 6.6	32.5	39.1	(2) 6.6	32.5	39.1
UNMANNED SUPPLY SPACECRAFT						
FERRY/SUPPLY LAUNCH ESCAPE SYSTEM	(2) 1.4	6.9	8.3	(2) 1.4	6.9	8.3
SPACECRAFT SPARES	17.0		17.0	18.4		18.4
SPACECRAFT LAUNCH SERVICE	20.8		20.8	21.7		21.7
SPACECRAFT PROPELLANT & TRANSPORTATION	1.7		1.7	1.8		1.8
GEMINI LAUNCH VEHICLES	(7)77.0		77.0	(7)77.0		77.0
SATURN LAUNCH VEHICLES	(2)60.0		60.0	(2)60.0		60.0
A.G.E. (FERRY & SUPPLY)		50.7	50.7		52.8	52.8
LAUNCH PAD CONVERSION		0.8	0.8		0.8	0.8
TRAINERS & SUPPORT		0.8	0.8		1.1	1.1
DEVELOPMENT FLIGHT TEST		94.0	94.0		143.4	143.4
TOTAL PROGRAM COST →			481.2			555.5

SCHEDULE C

	COMBINATION 1 (GLV / S-I) R	NR	TOTAL	COMBINATION 2 (GLV / S-I) R	NR	TOTAL
FERRY RE-ENTRY MODULES	(9)62.0	25.4	87.4	(9)65.2	30.0	95.2
FERRY ADAPTERS	(9)14.9	8.7	23.6	(9)18.8	17.1	35.9
SUPPLY MODULES	(2) 6.6	32.5	39.1	(2) 6.6	32.5	39.1
UNMANNED SUPPLY SPACECRAFT						
FERRY/SUPPLY LAUNCH ESCAPE SYSTEM	(2) 1.4	6.9	8.3	(2) 1.4	6.9	8.3
SPACECRAFT SPARES	17.0		17.0	18.4		18.4
SPACECRAFT LAUNCH SERVICE	20.8		20.8	21.7		21.7
SPACECRAFT PROPELLANT & TRANSPORTATION	1.7		1.7	1.8		1.8
GEMINI LAUNCH VEHICLES	(7)77.0		77.0	(7)77.0		77.0
SATURN LAUNCH VEHICLES	(2)50.0		50.0	(2)50.0		50.0
A.G.E. (FERRY & SUPPLY)		50.7	50.7		52.8	52.8
LAUNCH PAD CONVERSION		0.8	0.8		0.8	0.8
TRAINERS & SUPPORT		0.8	0.8		1.1	1.1
DEVELOPMENT FLIGHT TEST		89.0	89.0		133.4	133.4
TOTAL PROGRAM COST →			466.2			535.5

SCHEDULE D

	COMBINATION 1 (GLV / GLV) R	NR	TOTAL	COMBINATION 2 (GLV / GLV) R	NR	TOTAL	COMBINATION 3 (GLV / GLV) R	NR	TOTAL
FERRY RE-ENTRY MODULES	(8)55.1	25.4	80.5	(8)57.9	30.0	87.9	(8)55.1	25.4	80.5
FERRY ADAPTERS	(8)13.2	8.7	21.9	(8)16.7	17.1	33.8	(8)13.2	8.7	21.9
SUPPLY MODULES									
UNMANNED SUPPLY SPACECRAFT	(6)23.3	43.4	66.7	(6)33.9	48.8	82.7	(6)33.9	48.8	82.7
FERRY/SUPPLY LAUNCH ESCAPE SYSTEM									
SPACECRAFT SPARES	18.6		18.6	22.0		22.0	20.7		20.7
SPACECRAFT LAUNCH SERVICE	27.2		27.2	28.0		28.0	27.2		27.2
SPACECRAFT PROPELLANT & TRANSPORTATION	1.9		1.9	2.2		2.2	2.1		2.1
GEMINI LAUNCH VEHICLES	(11)			(11)			(11)		
SATURN LAUNCH VEHICLES	121 0		121.0	121.0		121.0	121.0		121.0
A.G.E. (FERRY & SUPPLY)		62.0	62.0		64.6	64.6		62.5	62.5
LAUNCH PAD CONVERSION		1.9	1.9		1.9	1.9		1.9	1.9
TRAINERS & SUPPORT		2.3	2.3		2.4	2.4		2.3	2.3
DEVELOPMENT FLIGHT TEST		20.9	20.9		68.3	68.3		20.2	20.2
TOTAL PROGRAM COST →			424.9			514.8			443.0

SCHEDULE E & F

	COMBINATION 1 (GLV / GLV) R	NR	TOTAL	COMBINATION 2 (GLV / GLV) R	NR	TOTAL
FERRY RE-ENTRY MODULES	(8)55.1	25.4	80.5	(8)57.9	30.0	87 9
FERRY ADAPTERS	(8)13.2	8.7	21.9	(8)16.7	17.1	33 8
SUPPLY MODULES						
UNMANNED SUPPLY SPACECRAFT	(4)15.6	43.4	59.0	(4)22.6	48.8	71 4
FERRY/SUPPLY LAUNCH ESCAPE SYSTEM						
SPACECRAFT SPARES	16.9		16.9	19.6		19 6
SPACECRAFT LAUNCH SERVICE	24.0		24.0	24.8		24 8
SPACECRAFT PROPELLANT & TRANSPORTATION	1.7		1.7	2.0		2.0
GEMINI LAUNCH VEHICLES	(10)			(10)		
SATURN LAUNCH VEHICLES	110.0		110.0	110.0		110.0
A.G.E. (FERRY & SUPPLY)		47.1	47.1		49.7	49.7
LAUNCH PAD CONVERSION		1.3	1.3		1.3	1.3
TRAINERS & SUPPORT		2.3	2.3		2.4	2 4
DEVELOPMENT FLIGHT TEST		20.9	20.9		68.3	68 3
TOTAL PROGRAM COST →			385.6			471 2

NOTES:

1. R = RECURRING COSTS, NR = NONRECURRING COSTS, GLV = GEMINI LAUNCH VEHICLE, S-1 = SATURN 1 LAUNCH VEHICLE, S-IB = SATURN IB LAUNCH VEHICLE, F-1 – FERRY (NOSE DOCK – NOSE MOOR), F-2 = FERRY (NOSE DOCK – SIDE MOOR), F-3 = FERRY (AFT DOCK – AFT MOOR), FS-1 FERRY SUPPLY (NOSE DOCK – NOSE MOOR), FS-2 = FERRY/SUPPLY (NOSE DOCK-SIDE MOOR), FS-3 = FERRY/SUPPLY (AFT DOCK – AFT MOOR), US-1 = UNMANNED SUPPLY (STRIPPED GEMINI), US-2 = UNMANNED SUPPLY (SPECIFIC DESIGN).
2. QUANTITY OF ARTICLES SHOWN IN ( ).

* BACK-UP VEHICLES SHOWN ARE QUANTITIES REQUIRED TO OBTAIN OVER-ALL MISSION RELIABILITY OF .90.

FIGURE 15.3-1

15.3 (Continued)

the program costs are shown only for selected combinations. However, at least two different Ferry configurations and two different supply configurations are shown for each schedule. In any schedule, the combinations of vehicles shown represent the least expensive combination (Combination 1) and the most expensive combination (Combination 2) which satisfies the 0.90 reliability requirement.

The spacecraft unit cost for each configuration is shown in Table 15.3-1. The Gemini spacecraft (parachute version) unit cost is used as a base point from which the cost is derived. The unit cost for any configuration is obtained by adding the net cost of each change to the basic cost.

Spacecraft non-recurring costs are those items required on a one-time basis and are composed primarily of the following items: engineering, tooling, reliability and qualification testing. Both contractor and sub-contractor costs are included. The costs were determined by a comparative analysis of Gemini costs and include only those items requiring new or additional development.

Launch vehicle costs are taken from Reference 15.3-1. The costs are assumed to include items such as spares, launch service, and propellants.

Aerospace Ground Equipment (AGE) costs include both in-plant and launch facility AGE. The AGE required at the launch facility includes equipment for the industrial facility, static firing tests, and liquid test facility. The cost of AGE for these facilities depends upon the number of spacecraft in-flow. For Schedule D, four spacecraft are in-flow at once. In all other schedules, three spacecraft are in-flow simultaneously. For the new pad facilities one set of AGE is required for each new pad, either Gemini Launch Vehicle (GLV) or Saturn.

For launch pad conversion it is assumed that all new pads for GLV are already Titan II rated. The costs shown for this item are those required to convert Titan II pads to GLV pads, and are based upon actual costs incurred for

TABLE 15.3-1

SPACECRAFT UNIT COST
(MILLIONS OF DOLLARS)

	F-1 OR F-2		F-3		FERRY /SUPPLY MODULE		US-1		US-2		REFURBISHED US-1	
	NR	R	NR	R	NR	R	NR	R	NR	R	NR	R
COST OF BASIC PROJECTED GEMINI RE-ENTRY MODULE		6.61		6.61		—		6.61		6.61		4.15
COST OF BASIC PROJECTED GEMINI ADAPTER		1.58		1.58		1.58		1.58		1.58		1.58
TOTAL GEMINI		8.19		8.19		1.58		8.19		8.19		5.73
CHANGES:												
GENERAL STRUCTURE					9.1	+1.20	2.9	–.10	8.3	–.80	2.9	–.10
NOSE BEEF-UP	1.2	+.03										
RETROGRADE SYSTEM	.6	+.02	.6	+.02	.6	+.02	.9	–.07	.9	–.07	.9	–.07
HEAT SHIELD THICKNESS INCREASE	1.2	+.03	1.2	+.03								
DELETE HEAT SHIELD								–.11		–.11		–.11
ENVIRONMENTAL CONTROL SYSTEM	12.2	+.18	6.3	+.03	.4	+.03	2.9	–.21	2.9	–.21	2.9	–.21
ELECTRICAL SYSTEM	2.8	–.03	2.8	–.03	1.9	–.02	1.4	–.19	1.4	–.19	1.4	–.19
ADDITION OF SEPARATION ROCKETS	1.0	+.03	1.0	+.03	1.0	+.03						
DATA LINK FOR RENDEZVOUS BACK-UP	2.5	+.10	2.5	+.10								
STANDBY TEMPERATURE CONTROL EQUIPMENT	.3	+.03	.3	+.03								
DELETION OF R & D INSTRUMENTATION	.1	–.14	.1	–.14	.1	–.07	.2	–.46	.2	–.46	.2	–.46
HATCH IN HEAT SHIELD			4.5	+.24								
HATCH IN PRESSURE BULKHEAD			1.9	+.32								
DELETE NOSE DOCKING PROVISIONS			.1	–.09								
ADAPTER TUNNEL			11.3	+.35								
AFT DOCKING CONTROLS			2.0	+.12								
AFT DOCKING RING			.3	+.03								
DOCKING COMMAND RECEIVER					.3	+.03						
ORBIT AND MANEUVERING SYSTEM					19.1	+.51						
SELF INJECTION SYSTEM							27.4	+.79	27.4	+.79	27.4	+.79
GUIDANCE, CONTROL, & R.C.S.							2.3	–.57	2.3	–.57	2.3	–.57
COMMUNICATIONS							.6	–.18	.6	–.18	.6	–.18
C.G. BALLAST REMOVAL								–.02		–.02		–.02
DELETE LANDING SYSTEM								–.13		–.13		–.13
DELETE RECOVERY SYSTEM								–.04		–.04		–.04
DELETE BIO-MED SYSTEM								–.09		–.09		–.09
DELETE RADAR, ADD TRANSPONDER							.6	–.12	.6	–.12	.6	–.12
DELETE CREW AND SURVIVAL EQUIPMENT								–.35		–.35		–.35
PROVISION FOR INTERACTION ON OTHER SYSTEMS, INCREASED ENVIRONMENTAL PROBLEMS, ETC.	12.2	+.10	12.2	+.10			4.2	+.01	4.2	+.01	4.2	+.01
TOTAL UNIT COST	34.1	8.54	47.1	9.33	32.5	3.31	43.4	6.35	48.8	5.65	43.4	3.89

NR = NONRECURRING
R = RECURRING

F-1 = FERRY SPACECRAFT NOSE DOCK – NOSE MOOR
F-2 = FERRY SPACECRAFT NOSE DOCK – SIDE MOOR
F-3 = FERRY SPACECRAFT AFT DOCK – AFT MOOR
FS-1 = FERRY SUPPLY SPACECRAFT NOSE DOCK – NOSE MOOR
FS-2 = FERRY SUPPLY SPACECRAFT NOSE DOCK – SIDE MOOR
FS-3 = FERRY SUPPLY SPACECRAFT AFT DOCK – AFT MOOR
US-1 = UNMANNED SUPPLY SPACECRAFT – STRIPPED GEMINI
US-2 = UNMANNED SUPPLY SPACECRAFT – SPECIFIC DESIGN

15.3 (Continued)

conversion of Pad 19 at AMR. A breakdown of these costs is given in Table 15.3-2.

For Saturn launch pads the costs of conversion reflect only Ferry Spacecraft additions (such as white room tent and Ferry/Supply maintenance platform). The estimated cost per Saturn launch pad for these additions is $60,000.

Costs of trainers and support are those associated with modifications to Gemini mission simulators, translation and docking simulators, and training aids. Requirements for new major simulators are not anticipated. Trainer and training aid support costs are computed on an 18 month basis.

TABLE 15.3-2
PAD 19 ADDITIONS FOR GLV (AFTER TITAN II MODIFICATIONS)

1. READY BLDG. FOR GEMINI		$ 33,000
2. WHITE ROOM (INCLUDING AIR CONDITIONING)		280,000
3. PERSONNEL ELEVATOR		30,000
4. ELECTRIC WIRING (EXTERIOR)		40,000
5. MISCELLANEOUS (GEMINI)		10,000
6. MAN RATED WATER SYSTEM		100,000
7. BRIDGE CRANE	WHITE ROOM	35,000
8. JIB CRANE	WHITE ROOM	12,000
9. PAVING EXCAVATION, DRAINAGE, GRADING POND FILL, DEMOLISH OLD TANK FARM NEW FENCES (1/2 OF TOTAL-GEMINI)		42,500
		582,500
ENGINEERING FOLLOW UP, ETC.		53,000
	TOTAL	$635,500

16. REFERENCES

2.1-1 NASA Request for Proposal, L-3408, "Gemini Spacecraft Study for Manned Orbiting Research Laboratory Ferry Missions," dated 18 April 1963.

4.3-1 NASA Request for Proposal, L-3315, "Comparative Studies and Preliminary Design of a Manned Orbital Research Laboratory (MORL) Concept," dated 15 April 1963.

7.1-1 Montgomery, James E., "Manned Control of Space Vehicle Docking," AAS Technical Session Paper, September 10, 1963.

7.1-2 McDonnell Specification Control Drawing, 52-52701, Revision F, "Model 133P Orbit Attitude and Maneuver System," dated 9 August 1963.

7.1-3 McDonnell Specification Control Drawing, 52-52700, "Model 133P Re-entry Control System," Revision G, dated 9 August 1963.

7.2-1 McDonnell Report No. 8951, "Project Gemini Reliability Model," dated 30 July 1962, Revised 1 November 1962.

7.3-1 McDonnell Report No. A179, "Ablative Heat Shield Hatch Development Program," dated 4 October 1963.

7.6-1 McDonnell Report No. 8518, "General Electrical Design Requirements for Electrical Equipment, and Equipment Utilizing Electrical Power for McDonnell Model 133P," dated 1 May 1962.

7.6-2 McDonnell Report No. 8951, "Project Gemini Reliability Model," dated 30 July 1962, Revised 1 November 1962.

8.1-1 Aerospace Corporation Report TOR-269(4126)-6, "Gemini Launch Vehicle Monthly Performance Report," dated 10 September 1963.

8.1-2 NASA Request for Proposal, L-3315, "Comparative Studies and Preliminary Design of a Manned Orbital Research Laboratory (MORL) Concept," dated 15 April 1963.

8.2-1 McDonnell Report No. 8616, "Structural Design Criteria, Gemini Spacecraft," dated 9 June 1962, Revised 1 July 1963.

8.2-2 McDonnell Report No. 9723, "Gemini Spacecraft Study for MORL Ferry Missions," dated 6 May 1963, Revised 7 July 1963.

8.2-3 McDonnell Report No. 9030, "Gemini Spacecraft Loads Report," to be published.

8.2-4 NASA Report M-AERO-PT-11-63, dated 8 May 1963.

8.2-5 NASA Report M-AERO-PT-12-63, dated 7 May 1963.

16. (Continued)

8.3-1 McDonnell Report No. 8611, "Gemini Spacecraft Performance Specifications," dated 20 February 1962, Revised 14 June 1962.

8.3-2 McDonnell Gemini Design Note No. 87, "Guidance and Control Mechanics," dated 4 March 1963.

8.3-3 McDonnell Gemini Design Note No. 65, "Guidance and Control Mechanics."

8.3-4 Heartz, R. A. and Jones, T. H., "Mistram and Rendezvous," Astronautics, July 1962.

8.5-1 McDonnell Report No. 8842, "Development Program for an Ablative Heat Shield Hatch," dated 10 May 1962.

8.5-2 McDonnell Report No. A179, "Ablative Heat Shield Hatch Development Program," dated 4 October 1963.

8.7-1 Arthur D. Little, Incorporated, Report C-63850, "Summary Report on a Study of the Blast Effect of the Saturn Vehicle," dated 15 February 1962.

8.8-1 Hess, W. N., "The Artificial Radiation Belt Made on 9 July 1962," Journal of Geophysical Research Vol. 68, No. 3, 667, dated 1 February 1963.

8.8-2 Hess, W. N., Private Communication (Memorandum), dated 25 June 1963.

8.8-3 Carter, R. E., Reines, F., Wagner, J. J. and Wyman, M. E., "Free Antineutrino Absorption Cross Section II, Expected Cross Section from Measurements of Fission Fragment Electron Spectrum," Physical Review, Vol. 113, No. 1, 280, dated January 1959.

8.8-4 Evans, R. D., "The Atomic Nucleus," McGraw-Hill, 1955, pp. 617, 627.

8.8-5 Price, B. T., Horton, C. C. and Spinney, K. T., "Radiation Shielding," Pergamon Press, 1957, pp. 72-76.

8.8-6 National Academy of Sciences - Space Science Board Committee 16, Working Paper, dated 26 March 1962.

9.2-1 Rosenwesser, H., "The Effects of Gamma Radiation on Explosives, ORNL-1720, 1955.

9.2-2 Kaufman, L. V. R., "The Effects of Nuclear Radiation on Explosives," Proceedings of the Royal Society (London), A246, 219, 1958.

9.2-3 Groocock, J. M., "The Effects of High Energy X-Rays and Pile Radiation on the Thermal Decomposition and Thermal Explosion of Lead Azide," Proceedings of the Royal Society (London), A246, 225, 1958.

9.2-4 Flanagan, Ted B., "The Effects of Reactor Irradiation upon Subsequent Thermal Decomposition of Lead Styphnate," Journal of Physics and Chemistry, 66.416 (1962).

16. (Continued)

11.1-1 McDonnell Report No. 233-K-200-20, "Project Gemini Weight and Balance Status, Number 20, " dated 31 August 1963.

13.2-1 NASA Request for Proposal, L-3408, "Gemini Space Station for Manned Orbiting Research Laboratory Ferry Missions," dated 18 April 1963.

15.3-1 NASA Request for Proposal, L-3408, "Gemini Space Station for Manned Orbiting Research Laboratory Ferry Missions," dated 18 April 1963.

B-1 Perkins, J. F., "Monte Carlo Calculation of Transport of Fast Electrons," Physical Review, Vol. 126, No. 5, p. 1781, dated 1 June 1962.

B-2 Wilson, R. R., "The Range and Straggling of High Energy Electrons," Physical Review, Vol. 87, No. 1, p. 100, dated 15 October 1951.

B-3 Marshall, J. S. and Ward, A. G., "Absorption Curves and Ranges for Homogenous β-Rays," Can. Journal of Research, Vol. 15, Sec. A, p. 39, dated January-December 1937.

B-4 Hereford, F. L. and Swann, C. P., "The Maximum Range of High Energy Electrons in Aluminum and Copper," Physical Review, Vol. 78, p. 727, dated 15 June 1950.

B-5 Motz, H. T. and Carter, R. E., "Artificial Radiation Belt Studies with a Fission β-Ray Source," Journal of Geophysical Research Vol. 68, No. 3, p. 657, dated 1 February 1963.

B-6 Evans, R. D., "The Atomic Nucleus," McGraw-Hill, 1955, pp. 617, 627.

B-7 Zerby, C. D., et al, "Tissue Dose Rate from Bremsstrahlung Radiation Behind an Idealized Apollo Vehicle Wall Exposed to Electrons in the Artificial Radiation Belt," Paper I-12 of Neutron Physics Division - Space Radiation Shielding Research Annual Progress Report for Period Ending 31 August 1962. ORNL-CF 62-10-29, Rev.

B-8 Price, B. T., Horton, C. C. and Spinney, K. T., "Radiation Shielding," Pergamon Press, 1957, pp. 72-76.

B-9 Goldstein, H. and Wilkins, Jr., J. E., "Calculations of the Penetration of Gamma Rays," Nuclear Development Associates Report NYO-3075 (NDA-15C-41), dated June 1954.

APPENDIX A

TWO-DAY RENDEZVOUS MISSION GEMINI

31 August 1963

The NASA Gemini Spacecraft which, with an Agena target spacecraft, will be used for in-space rendezvous and docking experiments is the base from which the Ferry Spacecraft for the MORL mission is developed. This version of the Gemini is designed for 2 day endurance and a maneuver-in-orbit capability of 700 fps.

<u>Spacecraft Design</u> - The Gemini Spacecraft (Figure A-1 and A-2) is a conical structure consisting basically of a re-entry module and an adapter section.

<u>Re-entry Module</u> - The re-entry module, which is patterned after the Mercury Spacecraft, is 144.2 inches long, and consists of a 90 inch diameter heat shield, the 20 degree conical crew and equipment section, and the 38.7 inch diameter rendezvous, recovery and reaction control equipment section. The crew and equipment section contains a pressurized crew compartment and several non-pressurized compartments for housing equipment. The re-entry control equipment section contains the major re-entry control system components. The rendezvous and recovery section contains the rendezvous radar equipment and the recovery system and is jettisoned with the drogue parachute after re-entry.

The re-entry heat protection design is similar to that proven on the Mercury Spacecraft. Basically it consists of beryllium shingles around the cylindrical section, Rene' 41 shingles on the conical body, and an improved ablative heat shield.

Escape off-the-pad and at altitudes below 70,000 feet during launch is accomplished by crew ejection seats. Above 70,000 feet the launch vehicle engines are shut down and the spacecraft is separated by salvo-firing the retrograde rockets.

Structural design details of the Re-entry Spacecraft include:

A. The pressure vessel consists of .010 smooth outer and .010 beaded inner titanium skins with flat, externally stiffened side and bottom panels.

SPACECRAFT PRE-LAUNCH CONFIGURATION

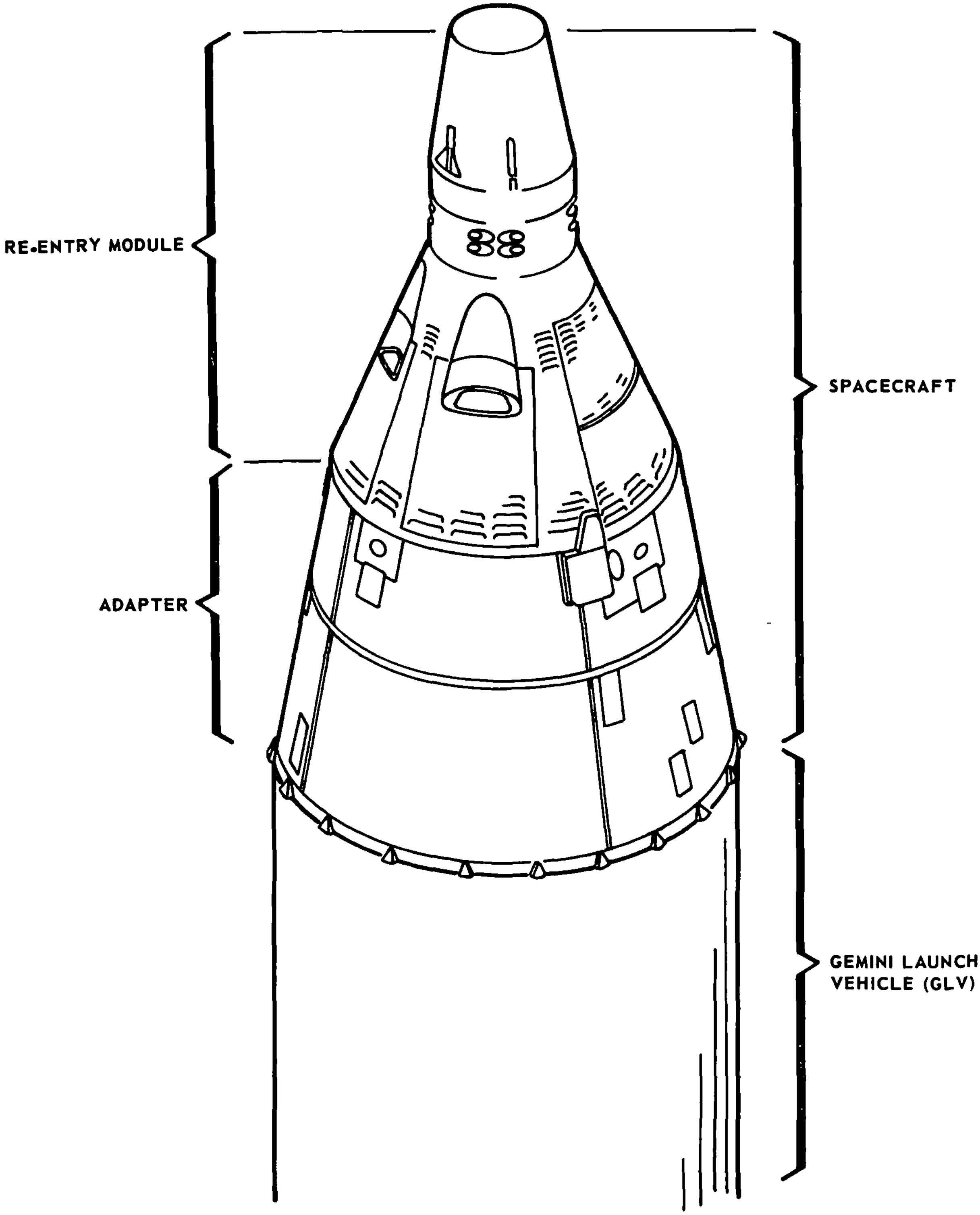


FIGURE A–1

INTERIOR ARRANGEMENT

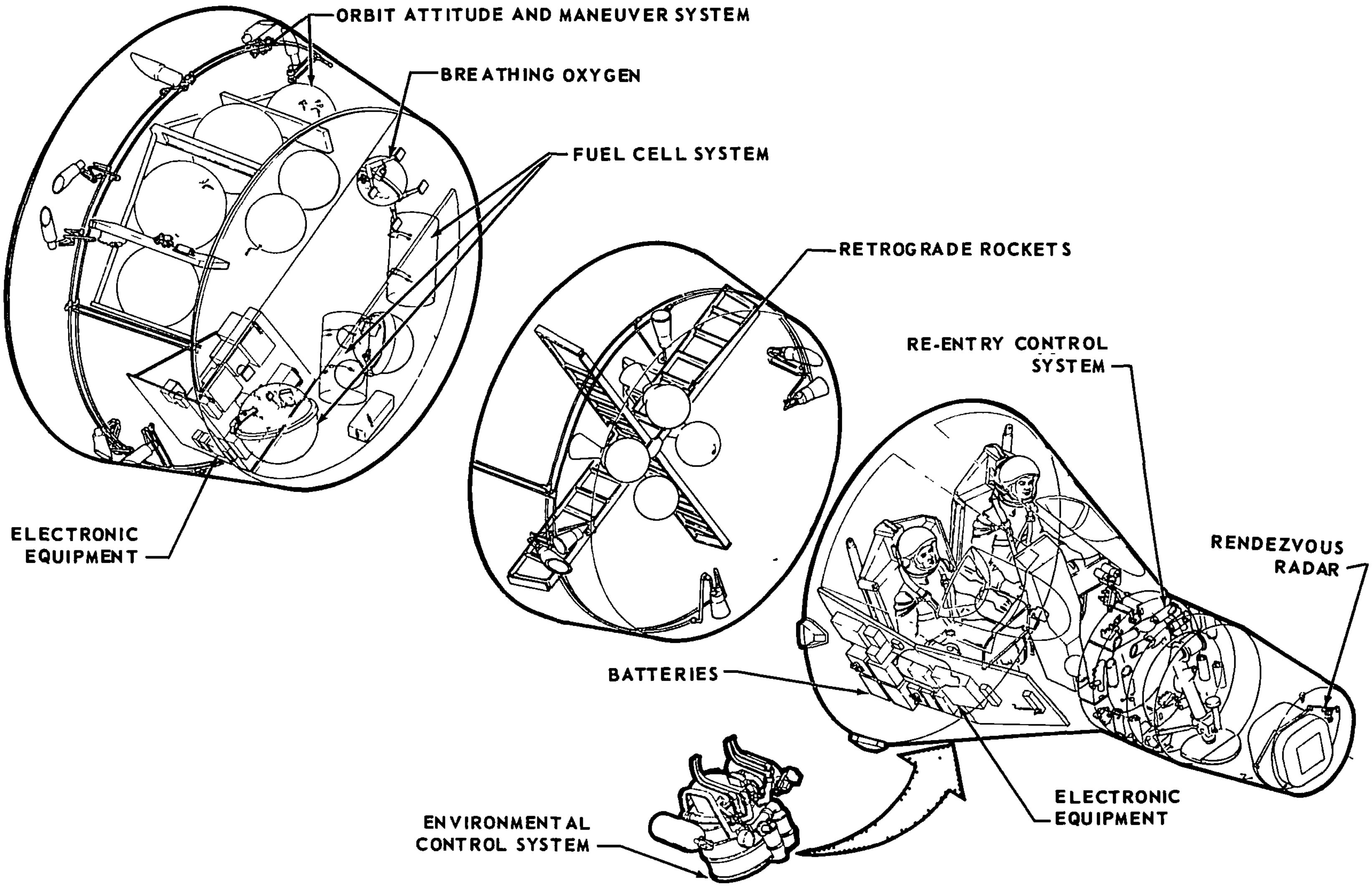


FIGURE A-2

B. Non-pressurized equipment compartments are located exterior to the flat panels.

C. The large pressure bulkhead is dome-shaped and consists of .010 smooth inner and .010 beaded outer titanium skins.

D. The small pressure bulkhead is flat and consists of .010 smooth outer and .010 beaded inner titanium skins.

E. Longitudinal stringers are titanium hat sections external to the structural conical skins and support the radiation cooled shingles.

F. Outer shingles are Rene' 41.

G. Large external access doors provide access to equipment located outside the pressurized cabin area.

H. Two large mechanically actuated hatches with integral windows are provided for ingress and egress. These must be opened prior to emergency ejection of astronauts.

I. A titanium box beam houses the paraglider control reels and cables and is located on centerline between the hatches. This box is covered with Rene' 41 shingles during orbit and re-entry.

Adapter - The adapter consists of the equipment section and the retrograde section. The equipment section is bolted to the launch vehicle forward oxidizer skirt. The equipment section contains major components of the fuel cell system, the Orbit Attitude and Maneuver System (OAMS), the major equipment cooling components, and the primary oxygen supply of the Environmental Control System (ECS). The retrograde section contains the retrograde rockets and some components of the equipment cooling system. At orbit insertion the adapter, which is built as a single structural unit, is severed at the launch vehicle attachment ring by shaped charges. Prior to retrograde the adapter is cut between the retrograde section and the equipment section, and the equipment section is jettisoned.

Structural design details of the adapter include:

A. The retrograde module is 35 inches long. The structural shell is also a space radiator. Stringers are extruded HM 31 magnesium with an integral tube for cooling fluid. The skin is .032 HK 31 magnesium.

B. The equipment module is 53 inches long, has the same construction as the retrograde module, and is a space radiator.

C. The adapter is attached to the launch vehicle by 20 bolts through the flange of a mating ring which also contains a shaped charge for separation. Attachment to the re-entry spacecraft is by three tension straps which can be severed by flexible linear shaped charges.

Navigation, Guidance and Control - Navigation, guidance and control of the Gemini spacecraft is accomplished by the astronaut using Time Reference, Inertial Guidance, Horizon Sensor, Rendezvous Radar, Attitude Control and Maneuver Electronics, and Propulsion systems. Provisions have been made in all related equipment for mechanizing automatic modes with attitude control inputs from radar, computer or platform. Provisions have also been made for incorporating an attitude hold feature.

The time reference system provides sequential signals of present time, elapsed time, and time-to-retrofire to the computer and the astronauts throughout the mission. The horizon sensor system provides outputs to align the inertial platform to the local earth vertical and provides a two-axis reference to the attitude control electronics during periods when the platform is not operating. Spacecraft-to-target bearing and range information from the rendezvous radar system is used by the inertial guidance system to provide rendezvous maneuver command signals for display to the astronaut. The attitude control and maneuver electronics system converts control signals to automatically command the propulsion system; or display information may be utilized directly by the astronauts to fire the appropriate

thrusters. This group of systems, with astronaut participation, provides guidance and control of the spacecraft through the orbital, retrograde and re-entry phases.

Rendezvous Radar - The rendezvous radar system (Figure A-3) is a cooperative radar which facilitates rendezvous and docking with the Agena target. The system consists of an interrogator in the Gemini spacecraft and a transponder in the Agena target vehicle. Each of the above units is a receiver-transmitter. Target range and bearing data is provided to the computer which then computes range, range rate, elevation, and azimuth data for display to the astronaut. This information is used to bring the spacecraft in close proximity to the Agena. When the range decreases to approximately 100 feet, the radar angular measurements become inaccurate and the astronaut must control the spacecraft either semi-automatically or manually. At 20 feet the radar drops out of the control function and the astronaut assumes manual control and visually completes the docking procedure. The rendezvous radar system sequence is shown in Figure A-4.

Communications, Command and Control - The communications, command and control system consists of two modes of voice communication, a Digital Command subsystem, two modes of coded identification beacons, an Acquisition Aid Beacon, a coded Recovery Beacon, three telemetry transmitters, and associated antennas and matching devices. Units of the communication system are located throughout the spacecraft.

UHF Voice Transmitter/Receiver - Two identical UHF voice transmitter/receivers are utilized to provide two-way voice communication between the spacecraft and ground stations. Both transceivers are located in the spacecraft right equipment bay.

HF Voice Transmitter/Receiver - Two identical HF voice transmitter/receivers are utilized to provide two-way voice communication between the spacecraft and ground stations. One HF transceiver is located in the spacecraft right equipment bay and the other is located in the adapter equipment section.

RENDEZVOUS RADAR SYSTEM

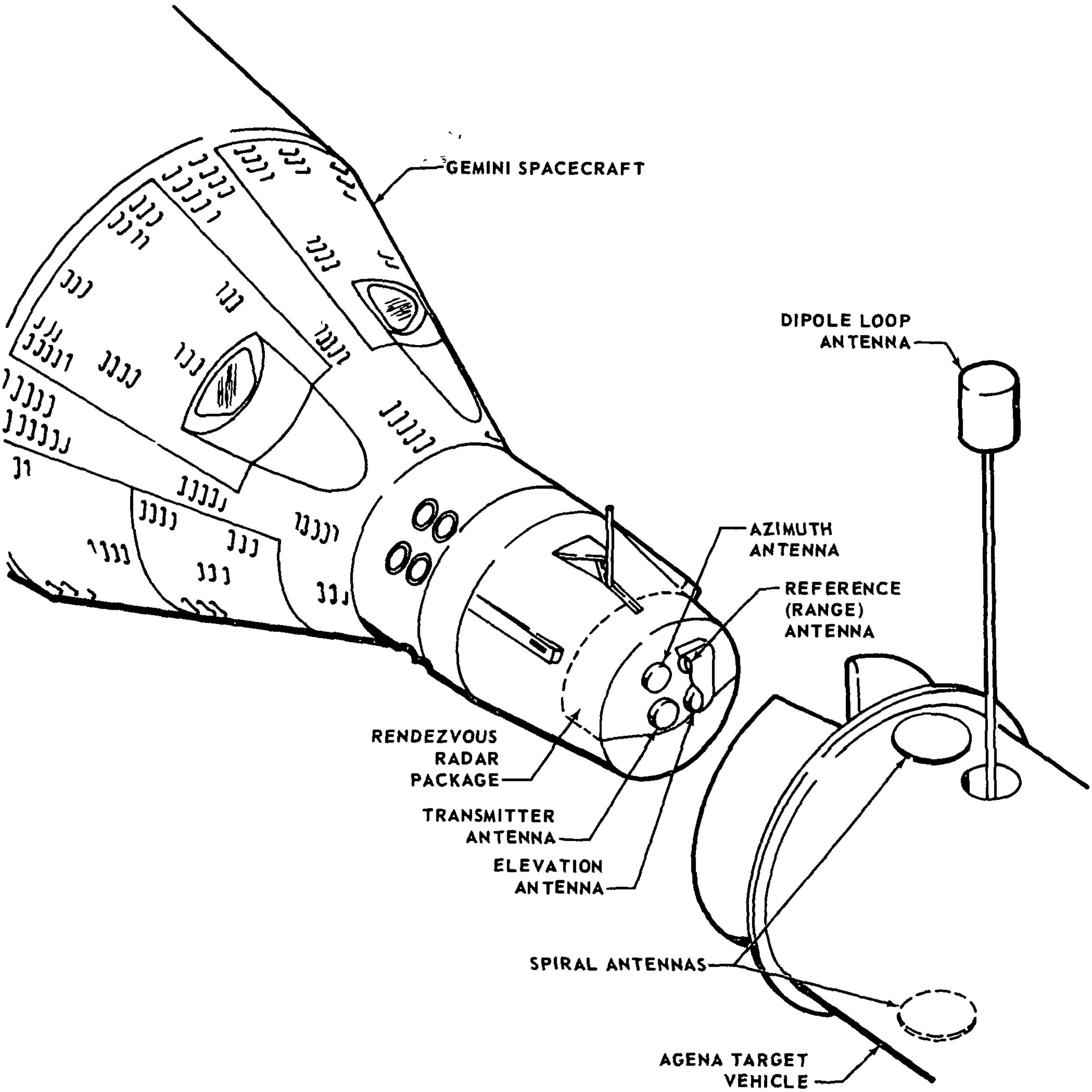


FIGURE A-3

RENDEZVOUS RADAR SYSTEM SEQUENCE

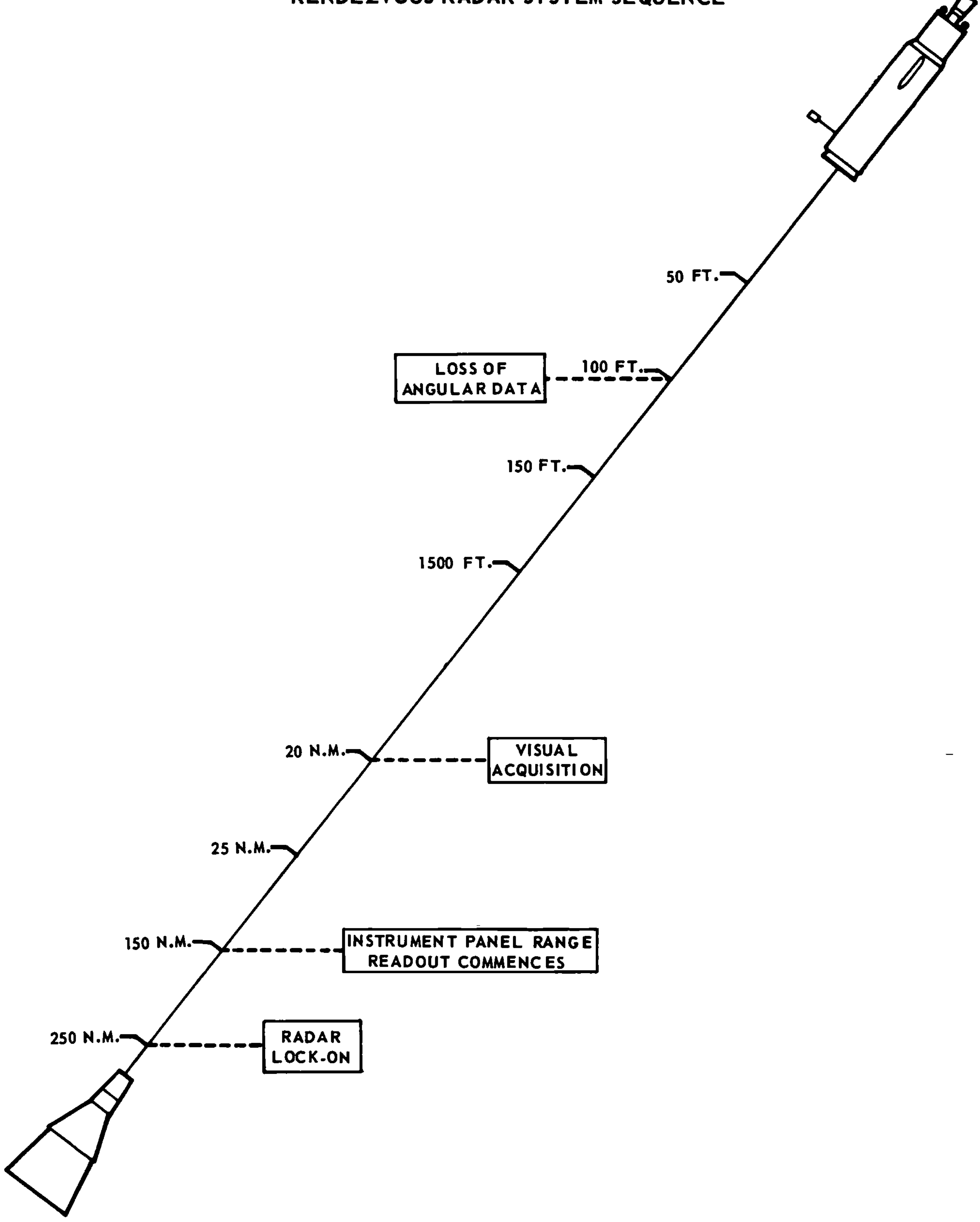


FIGURE A–4

Digital Command System - The digital command system (DCS) provides reception, conditioning and distribution to various spacecraft systems of ground-controlled, pulse-code-modulated transmissions.

The redundant receivers are FM superheterodyne type, operating in the UHF frequency spectrum. The output signal of the receiver is a phase-modulated audio tone.

The sub-bit detector demodulates the receiver output, separates the synchronization code from the data code and restores the data code to the original pre-transmission digital form.

The decoder receives both the synchronization and data codes from the sub-bit detector.

Telemetry Transmitters - Three telemetry transmitters, operating on separate carrier frequencies, provide FM modulated transmissions from the spacecraft to ground receiving stations. Data is derived from the spacecraft instrumentation system and can either be transmitted immediately or stored for later transmission.

Acquisition Aid Beacon - The acquisition aid beacon provides an acquisition signal for the ground tracking equipment. Normal operation is continuous wave, although phase modulation capabilities are incorporated.

UHF Recovery Beacon - The UHF recovery beacon, operating on the International Distress frequency of 243.0 MC, provides coded direction-finding information to recovery search units.

C & S Band Radar Beacons - The C & S band beacons are utilized to provide a reliable and precise tracking response to an interrogation signal through the reentry phase.

Propulsion - The Gemini spacecraft has two liquid bi-propellant propulsion systems to provide attitude control and maneuver capability. The Orbit Attitude and Maneuver System (OAMS) provides attitude control and maneuver capability from launch vehicle separation until spacecraft retrograde orientation. The OAMS is a

pressure-fed positive expulsion system operating on earth storable hypergolic propellants. Positive expulsion is accomplished by using flexible teflon bladders in all propellant tanks and pressurizing the tanks with a cold gas from high-pressure storage tanks. The OAMS is located in the equipment adapter and is jettisoned with the adapter shortly before retrograde rocket motor firing.

The Re-entry Control System (RCS) provides attitude control during retrograde and re-entry. Two completely separate redundant systems are provided and are similar in design to the OAMS. Each system is capable of performing all the control functions from retrograde through re-entry. The OAMS and RCS locations in the Gemini spacecraft are shown in Figure A-2.

The retrograde system imparts an impulse to the re-entry module to reduce the orbital velocity and establish a re-entry trajectory. The retrograde rocket motors also provide the required spacecraft separation from the launch vehicle in the event of a mission abort during launch at altitudes between 70,000 and 300,000 feet. The system consists of four Thiokol TE-385 motors mounted in the retrograde section of the adapter as shown in Figure A-2. These motors are ripple fired for orbital retrograde but must be salvo fired to escape during launch.

Environmental Control - The environmental control system supplies an oxygen environment for the pressure suits and spacecraft cabin and provides the necessary spacecraft and equipment temperature control (Figure A-5).

A primary oxygen system provides a 100% oxygen environment for launch and orbital flight. A secondary oxygen system supplies the oxygen during re-entry, and provides backup for the primary oxygen system. The egress oxygen system provides emergency oxygen for the astronauts should ejection from the spacecraft become necessary.

A closed-loop pressure suit circuit (Figure A-6) provides primary or secondary oxygen through the astronauts' pressure suits for breathing, ventilation, and press-

ENVIRONMENTAL CONTROL SYSTEM

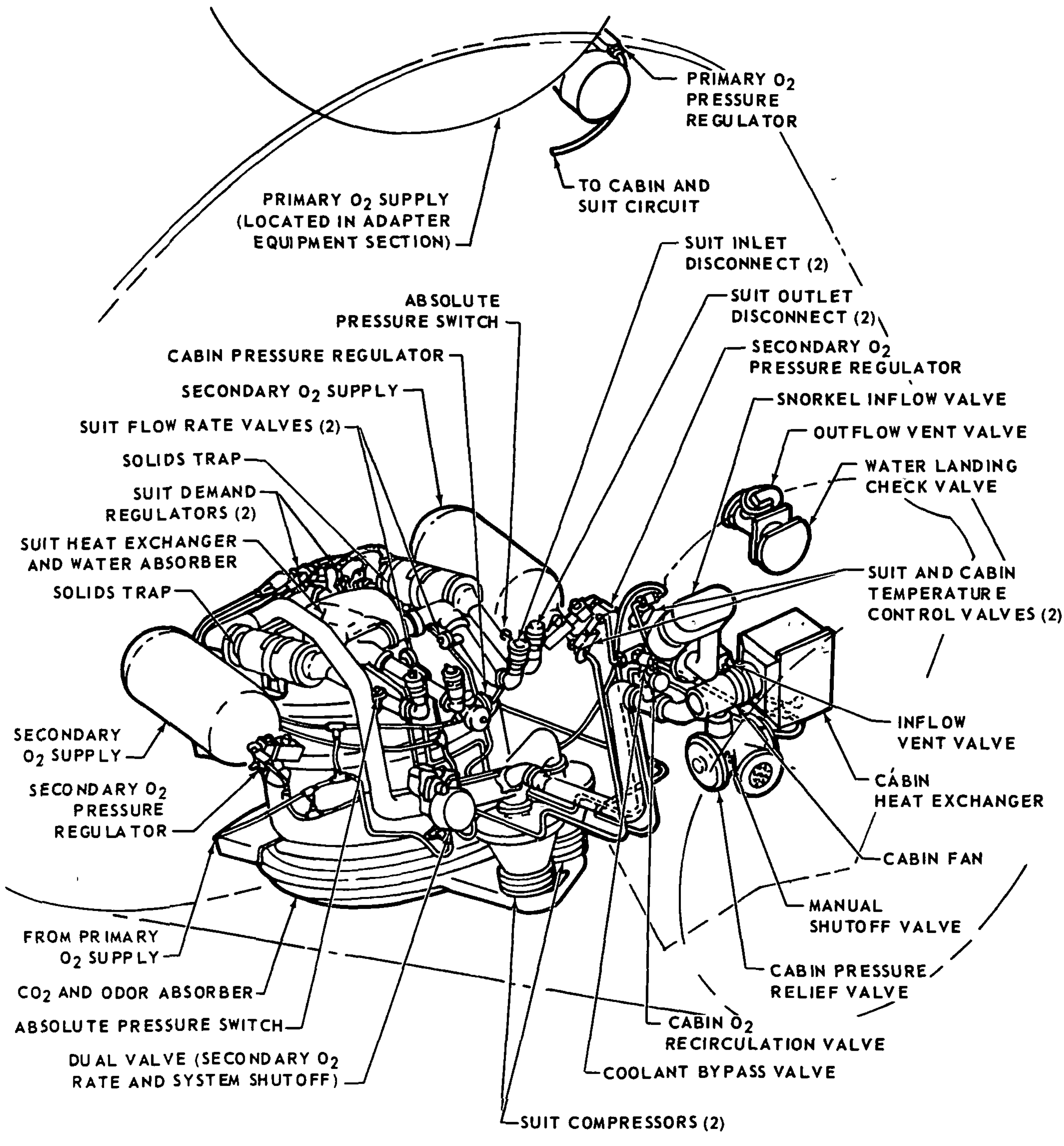


FIGURE A-5

ENVIRONMENTAL CONTROL SYSTEM SCHEMATIC

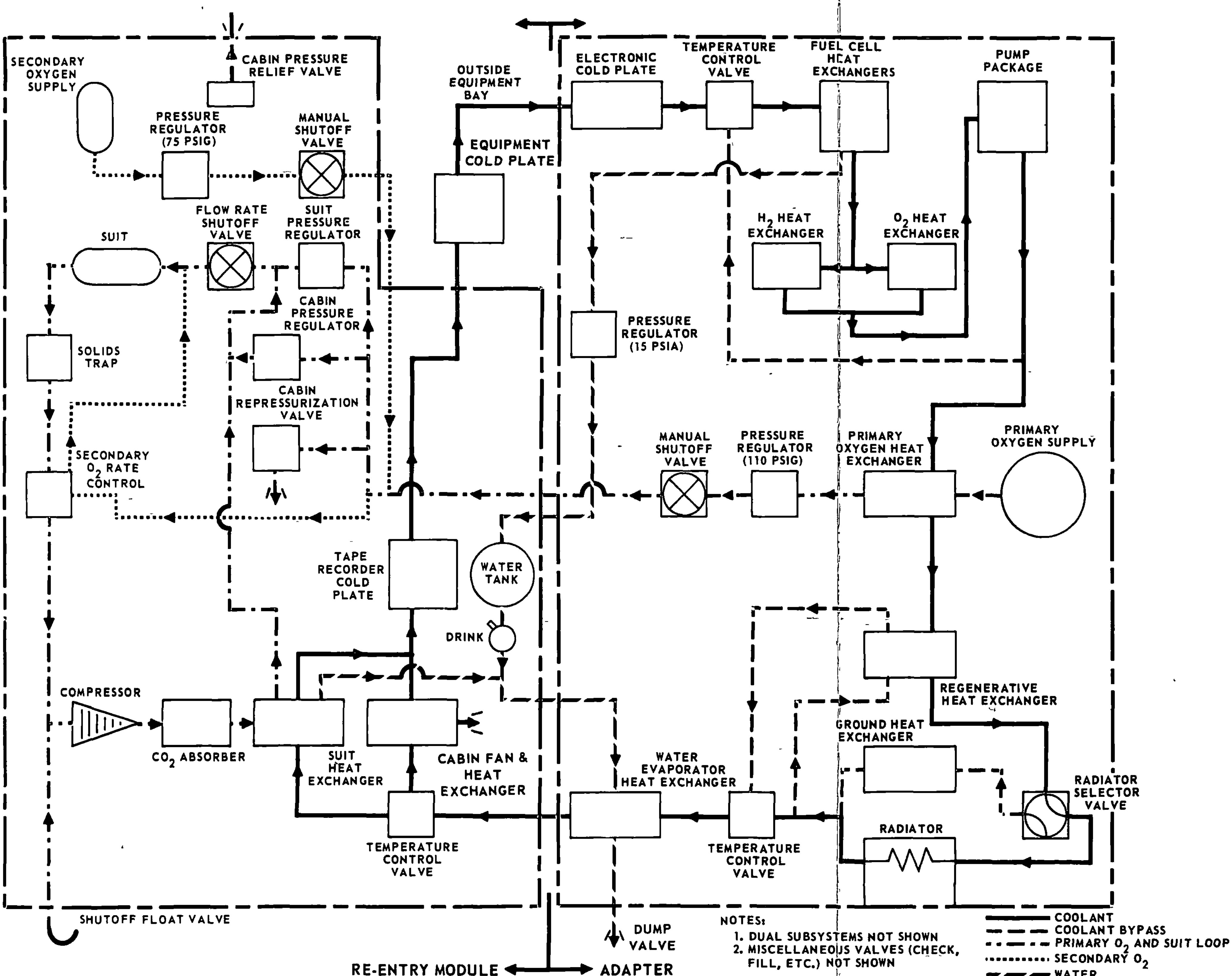


FIGURE A-6

urization. Electrically driven compressors circulate the gaseous flow through the pressure suit circuit, and the flow is controlled individually by manually-operated flow rate valves. Oxygen reconditioning units within the pressure suit circuit provide air purification, temperature control, and humidity control.

The spacecraft cooling system (Figure A-7) consists of two redundant temperature control coolant loops. Each coolant loop consists of a dual pump package, thermostatic and selector control valves, various type heat exchangers, a space radiator and the necessary plumbing required to provide a closed circuit. Either coolant loop will dissipate the maximum spacecraft heat loads.

Most of the equipment cold plates, cabin and suit heat exchangers are located in the re-entry module (Figure A-7). The pump packages, fuel cells, reactants heat exchangers, electronics cold plate, ground and regenerative heat exchanger,

COOLING SYSTEM

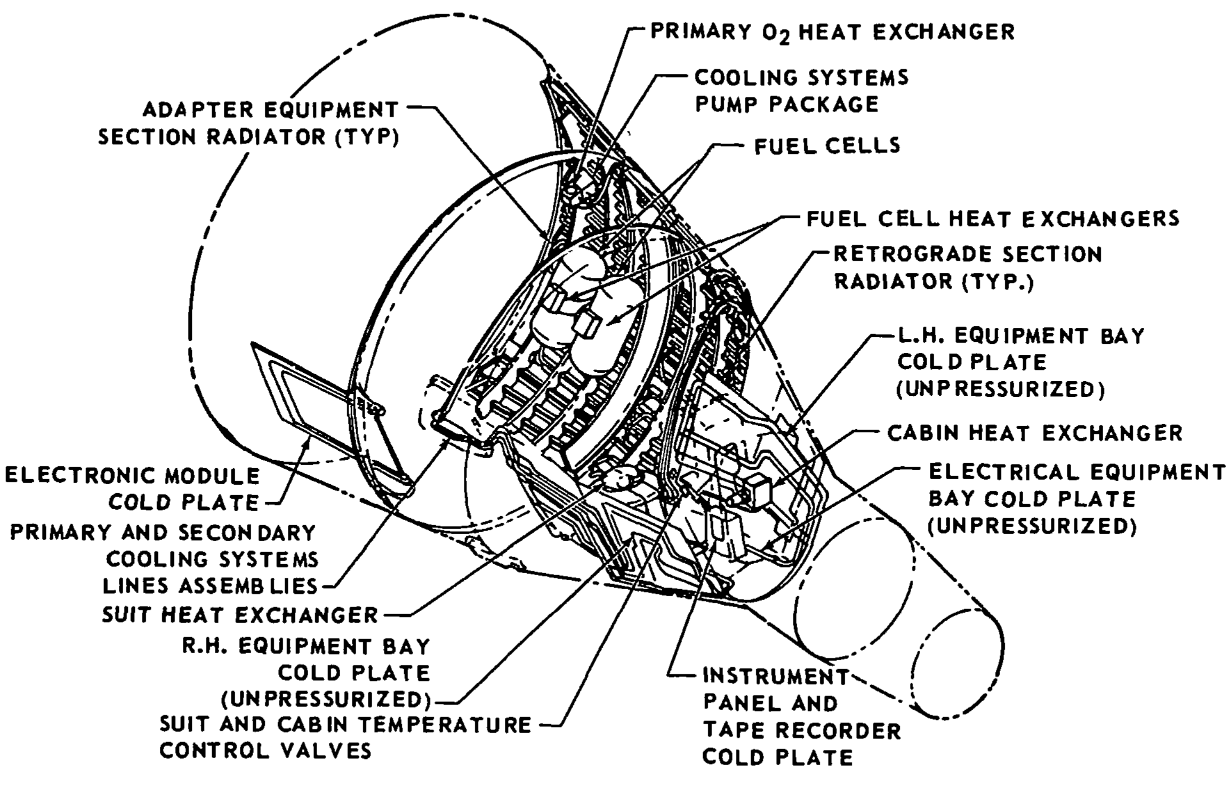


FIGURE A-7

primary O_2 heat exchanger, and the radiator panels are located in the adapter section. System manual controls are located on the astronauts' pedestal console and the control switches, warning lights and indicators are located on the center main instrument panel.

Power Supplies - The spacecraft electrical power system (Figure A-8) consists of fuel cells, batteries, provisions for utilizing ground power prior to launch, and the necessary switches and controls for proper distribution of DC power. The two fuel cell sections, rated at 1.05 KW each, are located in the equipment section of the adapter with their reactant supply system. Four main batteries (1080 WH each), for re-entry and post-landing power, and three squib and control batteries (360 WH each) are located in the RH equipment bay. The batteries are manually activated silver-zinc types. All circuits employ fuses or circuit breakers to maintain power system integrity.

ELECTRICAL POWER SYSTEM

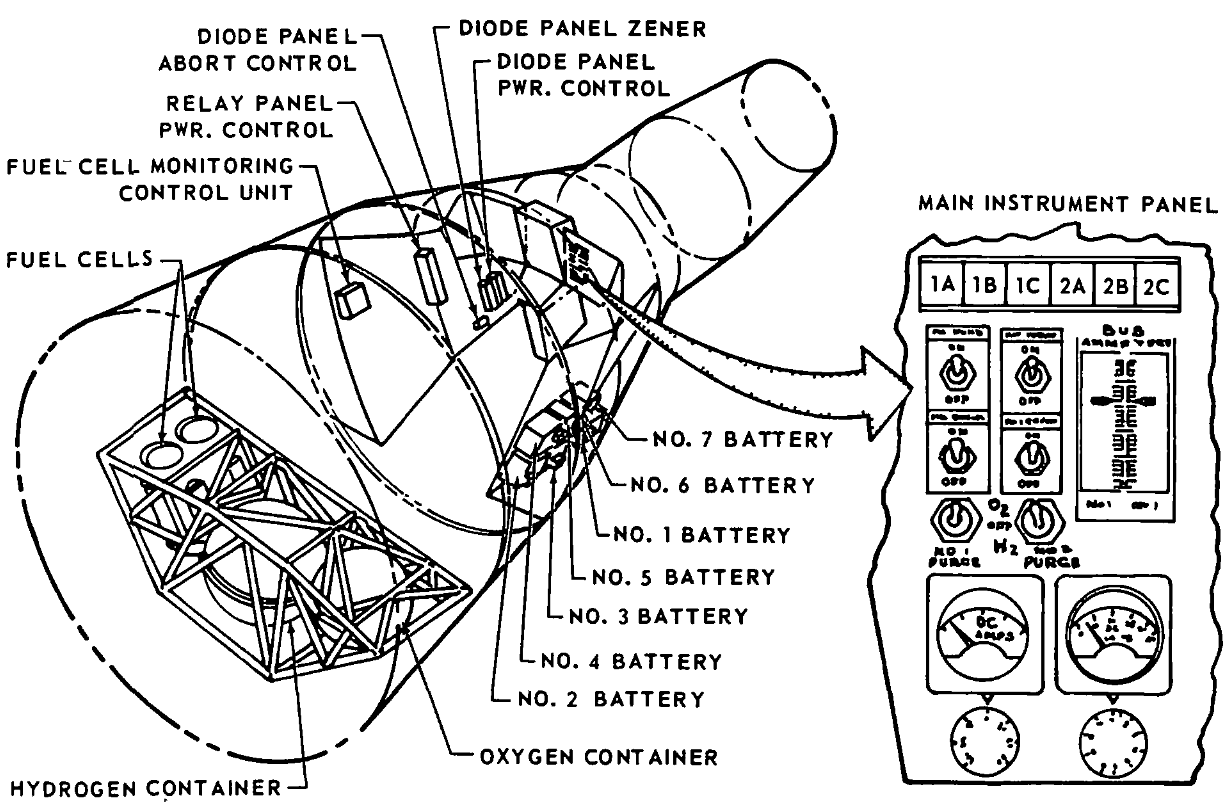


FIGURE A-8

The fuel cells and main battery outputs are fed to a common bus. The batteries serve as the primary source of electrical power for re-entry and post-landing portions of the mission, and as an emergency power source to permit re-entry in the event of complete fuel cell failure. For a partial fuel cell failure up to and including the loss of one complete section, the batteries may be used to augment the fuel cells during the relatively short periods of peak power demand.

The squib and control batteries are isolated from the main bus to prevent any voltage spikes being fed into critical electronics gear during operation of high current devices such as pyrotechnics, solenoids and relays. All pyrotechnic circuits, including the batteries, are redundant.

Landing Systems - Initial versions of the Gemini will utilize a Parachute Landing System as shown in Figure A-9. The parachute is an 84-foot diameter Ringsail. With this mode of recovery, a water landing is necessary to attenuate landing impact loads.

PARACHUTE LANDING SYSTEM

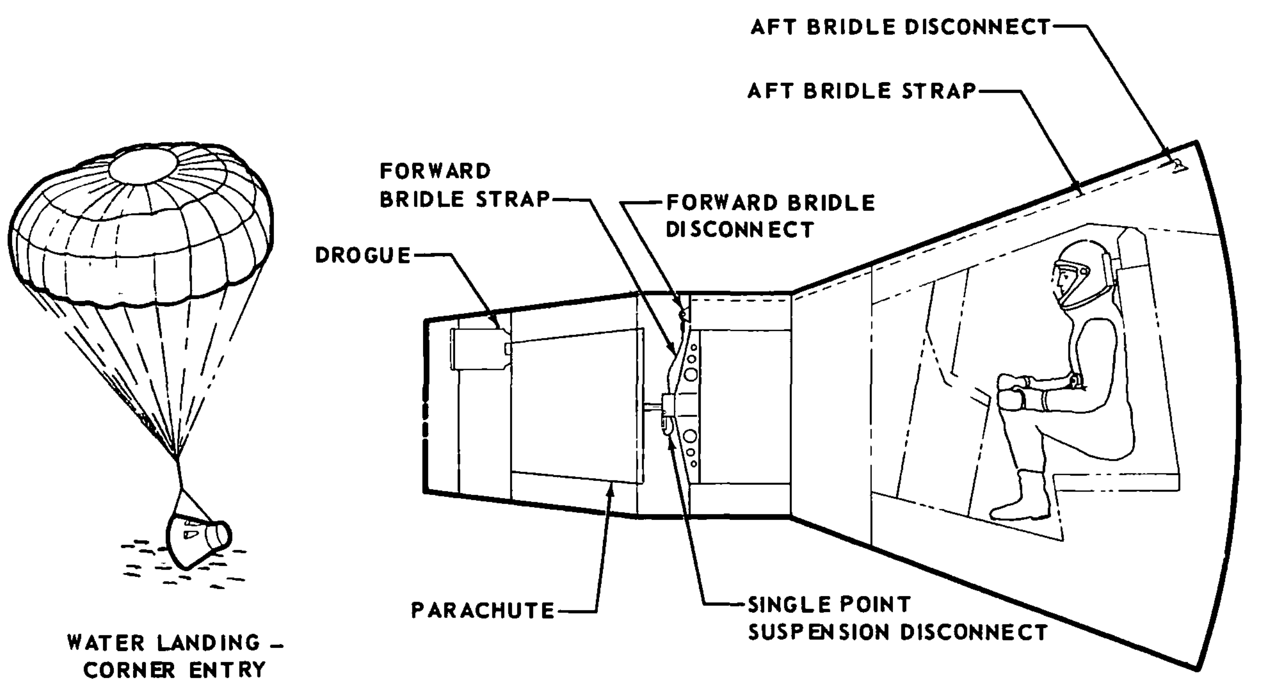


FIGURE A-9

Later versions of the Gemini are presently programmed to utilize a Paraglider Landing System as shown in Figure A-10. This system provides precise touchdown control and utilizes a tricycle skid landing gear.

A third landing system now under consideration is known as the Parasail and is shown in Figure A-11. This system uses a controllable parachute with a maximum L/D of 1.0. The Paraglider Landing Gear is retained, but since the Parasail cannot perform a flare maneuver landing, landing rockets are used to reduce the vertical velocity to within the design limits of the landing gear.

Flight Mechanics - To achieve rendezvous with a previously launched Agena target in a nearly circular 161 na. mi. orbit, the spacecraft is injected into an elliptical orbit having an 87 na. mi. perigee altitude and an apogee having the same altitude as the target vehicle (Figure A-12). Due to the shorter orbit period of the elliptical orbit, the spacecraft will "catch-up" to the target vehicle at a rate of approximately 5.5 degrees per orbit.

PARAGLIDER LANDING SYSTEM

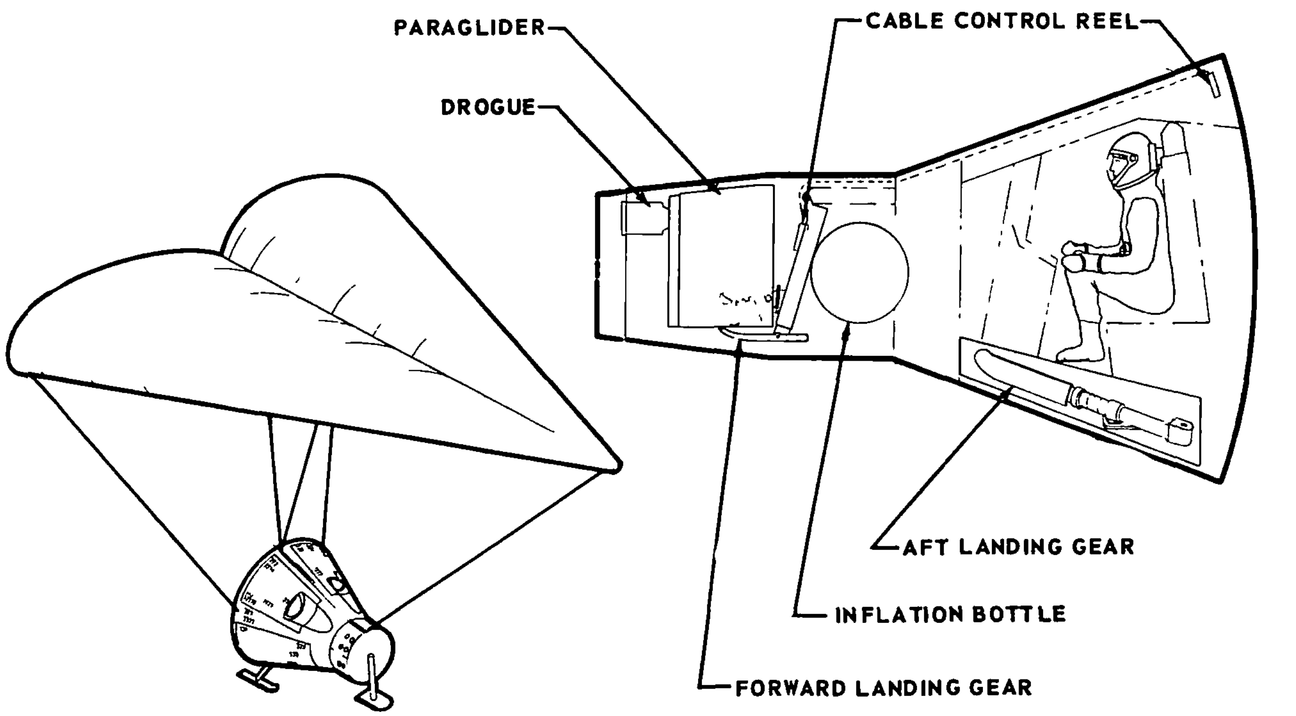


FIGURE A-10

PARASAIL LANDING SYSTEM

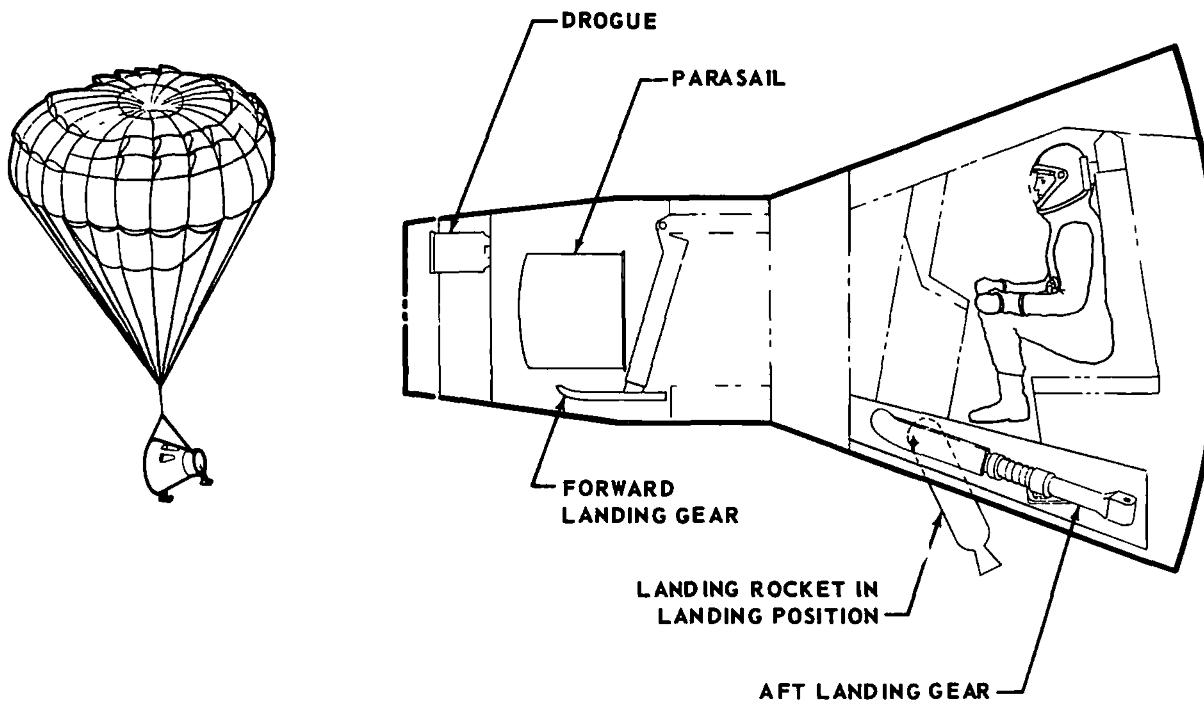
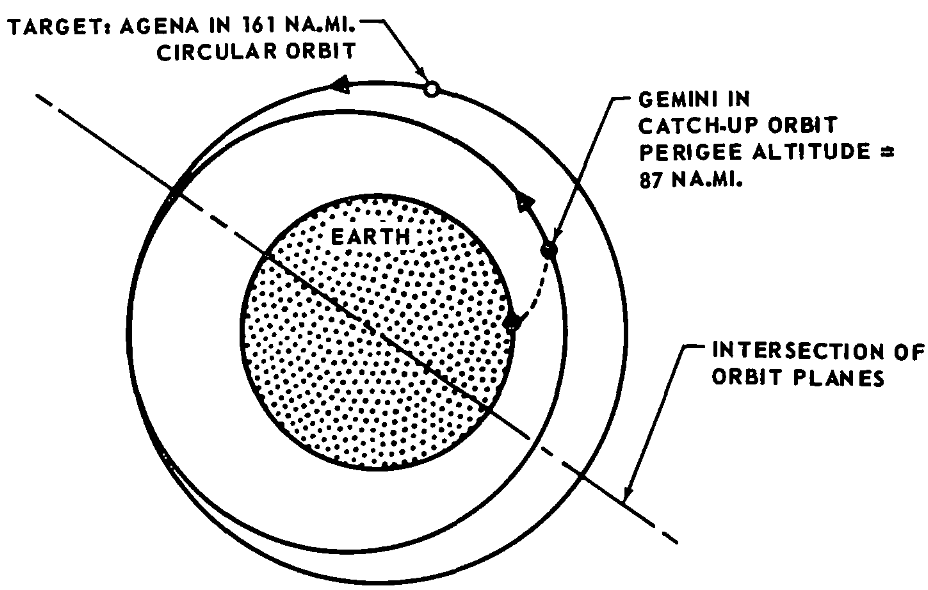


FIGURE A-11

GEMINI RENDEZVOUS WITH AGENA



- 700 FPS ΔV PROVIDED IN GEMINI FOR RENDEZVOUS.
- 5435 FPS ΔV AVAILABLE IN AGENA – CAN BE USED TO PROVIDE ADDITIONAL LAUNCH WINDOW FOR NON-OPTIMUM ORBIT INCLINATION OR TO ACCOMPLISH FAST CATCH-UP.

FIGURE A-12

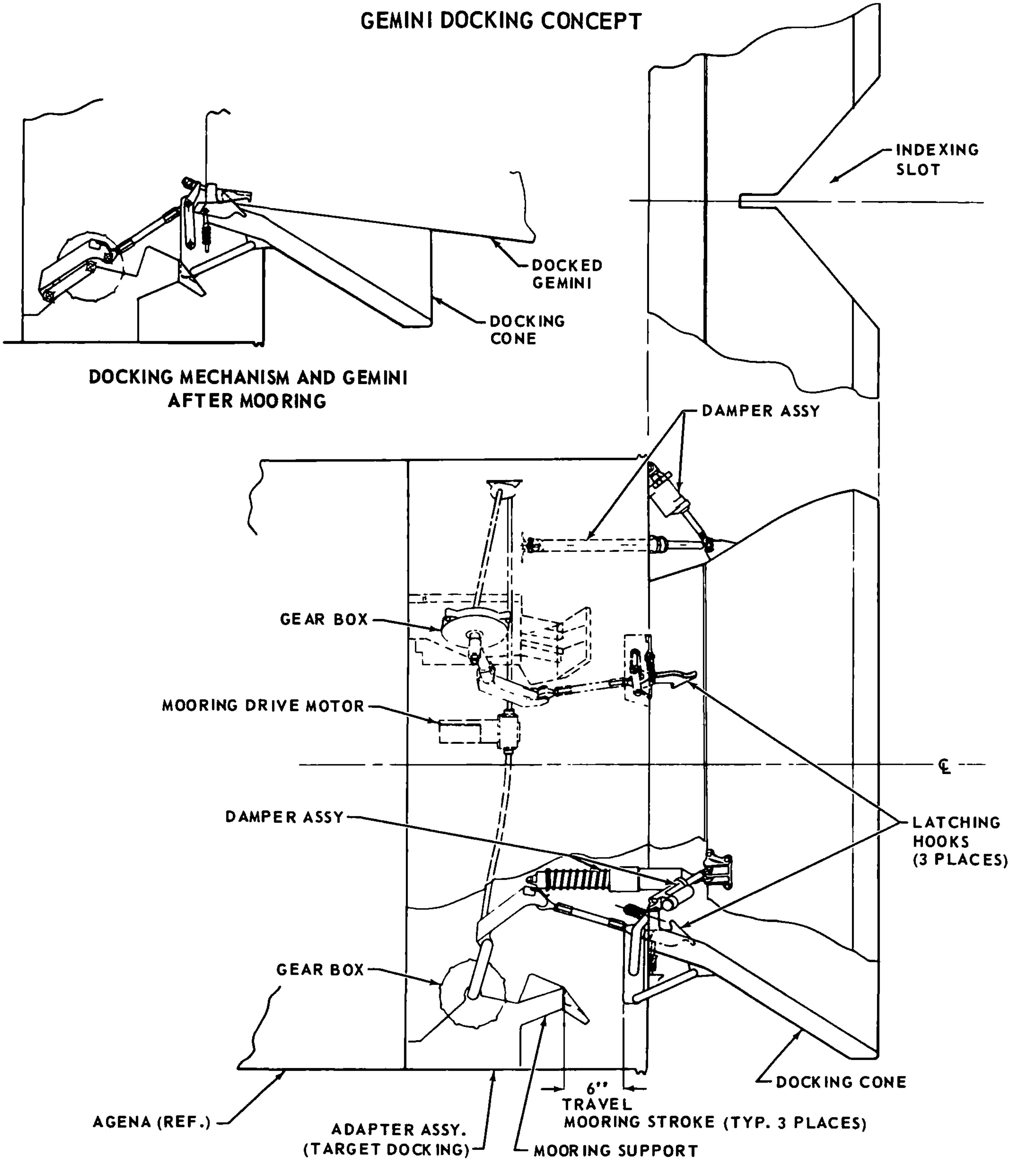
At a range of approximately 250 na. mi. from the target, ground radar control informs the astronaut to establish radar contact. After radar lock-on the astronaut initiates closed-loop collision course control. Required velocity increments are computed and corrective impulses are applied periodically during closed-loop collision course control.

When the spacecraft is within 10,000 feet of the target, the astronaut terminates the closed-loop collision course control and maneuvers the spacecraft manually in response to displays. At approximately 500 feet from the target, the astronaut maneuvers to align the spacecraft with the docking end of the target vehicle. The final docking maneuvers are performed by manual control while the astronaut observes the target vehicle through the window of the spacecraft. Docking is completed by mechanical latching (Figure A-13). For separation the spacecraft is unlatched and the astronaut initiates a thrust impulse which separates the spacecraft from the target vehicle. Following separation from the target vehicle, the spacecraft is readied for re-entry.

The spacecraft is positioned to the proper retrograde attitude, the equipment section is jettisoned, and firing of the retrograde rockets is initiated by the astronaut upon receipt of retro fire signal.

The re-entry phase begins with jettison of the retrograde section and orientation of the spacecraft to the re-entry attitude. A moderate lift capability is provided during re-entry by trimming the spacecraft to an angle of attack. The desired trim angle is attained by designing the spacecraft with the center-of-gravity offset approximately 1.5 inches from the longitudinal centerline. Since the C.G. position cannot be controlled during the mission, maneuverability is achieved by rolling the spacecraft to direct the lift vector in the desired direction or by maintaining a continuous roll rate for an effective zero lift re-entry trajectory. The lift thus generated provides a maneuver capability footprint approximately 80 na. mi. wide and 500 na. mi. long.

GEMINI DOCKING CONCEPT



DOCKING MECHANISM AND GEMINI BEFORE DOCKING

FIGURE A-13

The inertial guidance system measures the retrograde velocity increment and senses the lift and drag during descent. This information is used to determine position, velocity, and heading of the spacecraft. Closed loop control during re-entry is accomplished by continuous comparison of the computed zero lift and desired touchdown points. From this comparison, downrange and crossrange errors are determined. The required combinations of vertical and horizontal lift are logically selected by the computer and signals are provided to the attitude control system to control the spacecraft roll angle. By controlling the roll angle, the touchdown point is controllable to within a 4 mile radius at paraglider deployment.

APPENDIX B

METHOD OF CALCULATING RADIATION SHIELDING

Electrons - The bases of the electron dose rate attenuation calculations reported here are the published results of calculations and measurements of the transmission of monoenergetic electron beams through matter. (Reference B-1 through B-4). These measurements are used to calculate the number of fission spectrum electrons penetrating a slab shield and the associated dose. The dose attenuation factor, defined as the transmitted dose divided by the incident dose, is shown as a function of aluminum shield thickness in Figure 8.8-6.

The number of transmitted fission spectrum electrons differs from the dose attenuation curve of Figure 8.8-6 by less than 10%. The dose attenuation curve may therefore by compared with the transmission of fission spectrum electrons as measured by Motz and Carter (Reference B-5) for carbon, lucite, steel and lead. The measured values are shown in Figure 8.8-6 with the experimental counter wall correction factor included. The calculated curve for aluminum is in reasonable agreement with these measurements.

In performing the dose calculations for the spacecraft, an idealized shielding configuration is used. The spacecraft is represented by a series of surface segments of uniform shielding thickness. The method used to obtain this distribution for the Ferry is given in Section 8.8.2. Let ϕ_e (electrons/cm^2) be the omnidirectional flux of electrons encountered by the spacecraft. The number of electrons per unit area striking a shield segment (Figure B-1) is given by the electron current J. Assuming that the flux arrives isotropically the electron current is $J = \frac{\phi e}{4}$.

The number of particles reaching the inside surface per unit area is given approximately by $\phi_e/4 \cdot D/D_o\ (t)$, where t is the normal thickness of the segment and

SHIELD SEGMENT GEOMETRY

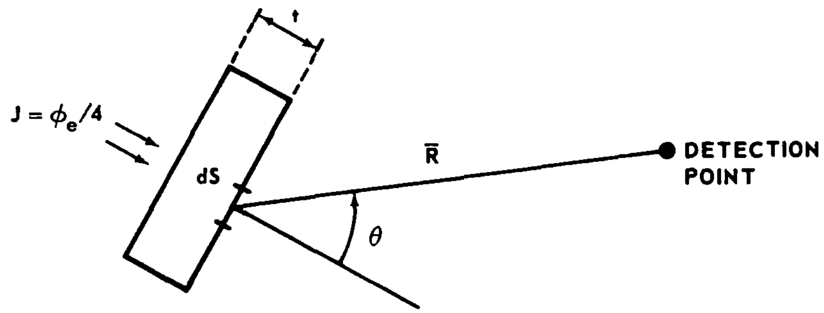


FIGURE B-1

D/D_o (t) is the dose rate attenuation factor (approximately equivalent to the transmission factor). Assuming an isotropic spatial distribution for the electrons leaving the inside surface (which is equivalent to a cosine surface distribution) the dose reaching the point of interest per unit detector area is:

$$\Delta D_e = f_e \frac{\phi_e}{4} \frac{D}{D_o} (t) \int \frac{dS \cdot \cos\theta}{\pi R^2} = f_e \, \phi_e \frac{D}{D_o} (t) \frac{\Delta\Omega}{4\pi} \qquad (1)$$

where the integration is over the inside surface of the segment, and ϕ is the angle formed by the normal to the surface and the radius R from the surface to the point of detection. $\Delta\Omega$ is the solid angle subtended at the receiver point by the shield segment, and f_e is the flux to dose conversion factor for the fission spectrum. Summing over all the shield segments the total dose is

$$D_e = \phi_e \, f_e \, \Sigma \frac{D}{D_o} (t_i) \frac{\Delta\Omega_i}{4\pi} \qquad (2)$$

<u>Bremsstrahlung</u> - The energy, ω, liberated in the form of gamma radiation when an electron is stopped in a shield is given approximately by (Reference B-6),

$$\omega = 7 \times 10^{-4} \, Z \, E_e^{\,2} \text{ (Mev)} \qquad (3)$$

Where Z is the atomic number of the shielding material and E_e is the incident electron energy in mev. The effective value of Z used in the calculations is 20.

Recent calculations at the Oak Ridge National Laboratory provide the energy spectra of bremsstrahlung produced in charring ablator material. (Reference B-7) The bremsstrahlung spectra are roughly independent of the Z number of the shielding material (Reference B-8) and it is assumed that the Oak Ridge spectra are appli-

cable for the Ferry Spacecraft. The Oak Ridge spectra, $B(E_e, \alpha)$, have units of photons per unit α. Where $\alpha = E_\gamma E_e$ is the ratio of gamma ray energy to the incident electron energy. These spectra are normalized to unit emitted gamma energy by the equation

$$B'(E_e, \alpha) = B(E_e, \alpha) / \int_0^{\alpha} B(E_e, \alpha)\, E_\gamma \, d\alpha . \quad \left(\frac{\text{PHOTONS/UNIT } \alpha}{\text{UNIT EMITTED ENERGY}} \right) \tag{4}$$

The bremsstrahlung produced is assumed to be radiated isotropically from a point at a depth within the shield equal to the maximum range of the incident electron (R max.). With this assumption the bremsstrahlung dose contributed by the shield segment of Figure B-1 may be written.

$$\Delta D_\gamma = \frac{\phi_e}{4} \int \frac{dS}{4\pi R^2} \int_0^{\infty} Y(E_e)\omega(E_e)\, dE_e \int_0^{\alpha} B'(E_e, \alpha)\, f(E_\gamma)\, B(E_\gamma, \mu_o r)\, e^{-\mu_o r}\, d\alpha \tag{5}$$

In this equation, $B(\mu_o r, E_\gamma)$ is the gamma ray dose build up factor for an isotropic point source (Reference B-9), $\mu_o o(E_\gamma)$ is the mass absorption coefficient for gamma rays of energy E_γ, and $r = t - R$ max for $t \geqslant R$ max. and $r = o$ for $t \leqslant R$ max., $Y(E_e)$ is the normalized curve fit to the fission spectrum and $f(E_\gamma)$ is the flux to dose conversion factor for gamma rays.

Performing the integrations over energy, with aluminum as the attenuating material, Equation (5) becomes

$$\Delta D_\gamma = 5.2 \times 10^{-12} \phi_e \int \frac{dS}{4\pi R^2} \tag{6}$$

Calculation of the value of the surface integral in Equation (6) for the detection point defined in Section 8.8.2 results in the value 1.28. The final expression obtained for the gamma ray dose is

$$D_\gamma = 6.7 \times 10^{-12} \phi_e \tag{7}$$

<u>Protons</u> - The dose per unit flux of protons of initial energy greater than 30 mev., $f_p(t)$, is shown in Figure 8.8-8 for aluminum slab shield thicknesses, t, from 1 to 10 gms/cm^2. The protons are assumed to be incident normally on the slab and only ionization loses are considered in calculating attenuation. The paths traversed by the protons in penetrating the shielding material are assumed to be

straight lines (straight through approximation).

Assuming an isotropic angular distribution for the proton flux, the dose received at the detector position from protons penetrating the shield segment of Figure B-1 is given by

$$\Delta D_P = \phi_P f_P (t/\cos\theta) \frac{\Delta\Omega}{4\pi} \qquad (8)$$

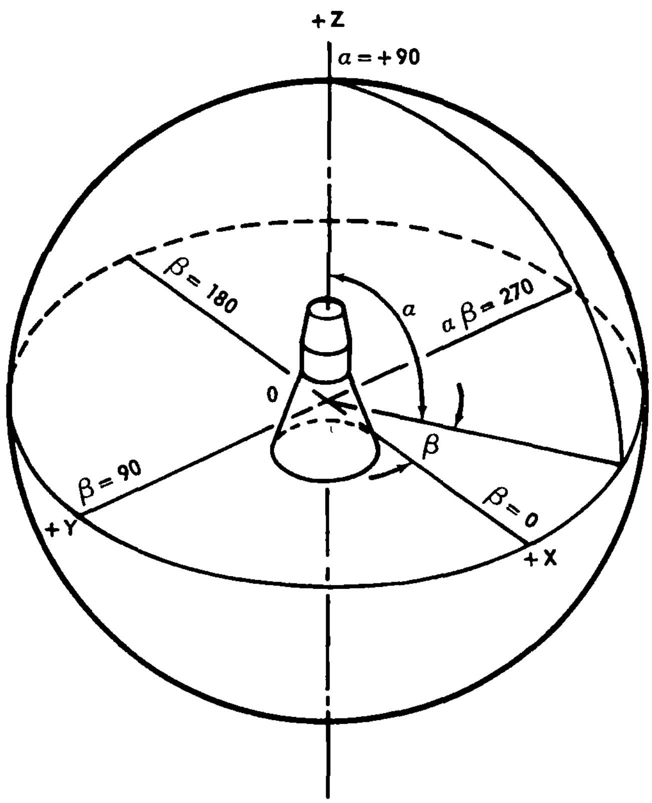
Where ϕ_p is the omnidirectional proton flux, $\Delta\Omega/4\pi$ is the fraction solid angle subtended at the detector position by the shield segment, and $t/\cos\theta$ is the "slant thickness" of the segment.

<u>Mass Distribution for Inherent Shielding</u> - The technique used to determine the inherent mass distribution of the Ferry Spacecraft for radiation shielding purposes is discussed in this section.

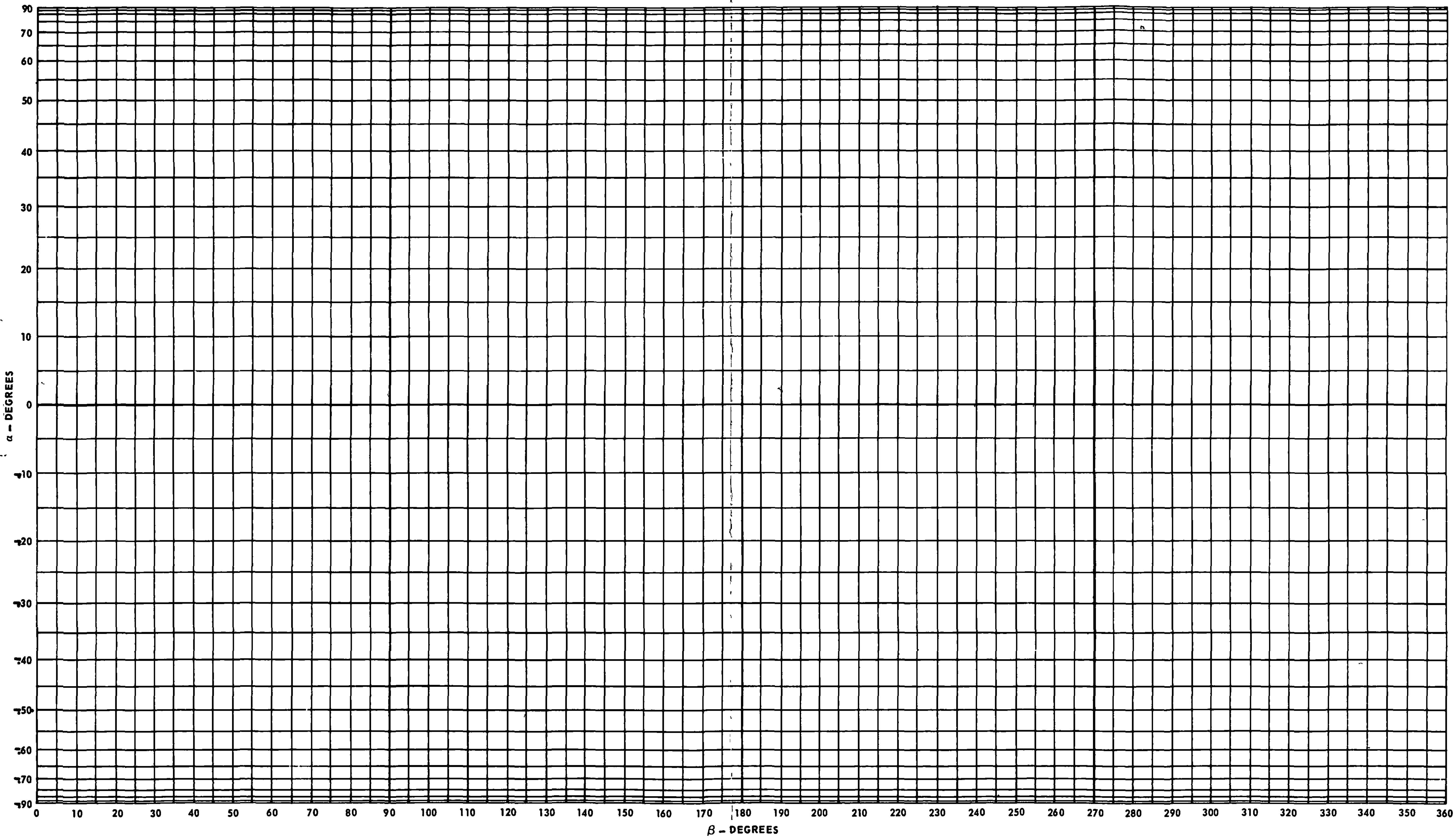
The technique differs from conventional methods primarily in that each item of structure and equipment is considered separately as opposed to combining large quantities of mass and using estimated equivalent shielding densities and arbitrary porosity factors.

In utilizing the item by item approach it is necessary to determine the portion of the total solid angle around the target point, i.e., point where the dose rate is to be determined, that is "shadowed" by each component of the spacecraft. This is accomplished by projecting (analytically) each component on a reference sphere (Figure B-1). The center of the reference sphere is located at the target point. Any point on the surface of the reference sphere is uniquely determined by two angular coordinates α, β, as shown in Figure B-1. The analytical projection procedure consists of first determining the α, β coordinates of rays passing from the target point through certain defining points of the component, for example, the corners of a black box. These coordinates are then plotted on an orthogonal α, β grid. This grid is shown in Figure B-2. The α scale is distorted such that the grid is equiareal, i.e., components plotted on the grid occupy the same percent

CO-ORDINATE SYSTEM
(INHERENT SHIELDING DISTRIBUTION)



ALPHA – BETA EQUIAREAL GRID



of the total grid area as the projection of these components on the reference sphere occupies of the total reference sphere surface area.

Areal densities (pounds per square foot) for each spacecraft component are determined for both the shortest distance through the part under consideration and the true distance measured along the radial ray from the reference point. These values are designated normal and slant height areal densities respectively and are noted on the α, β plots.

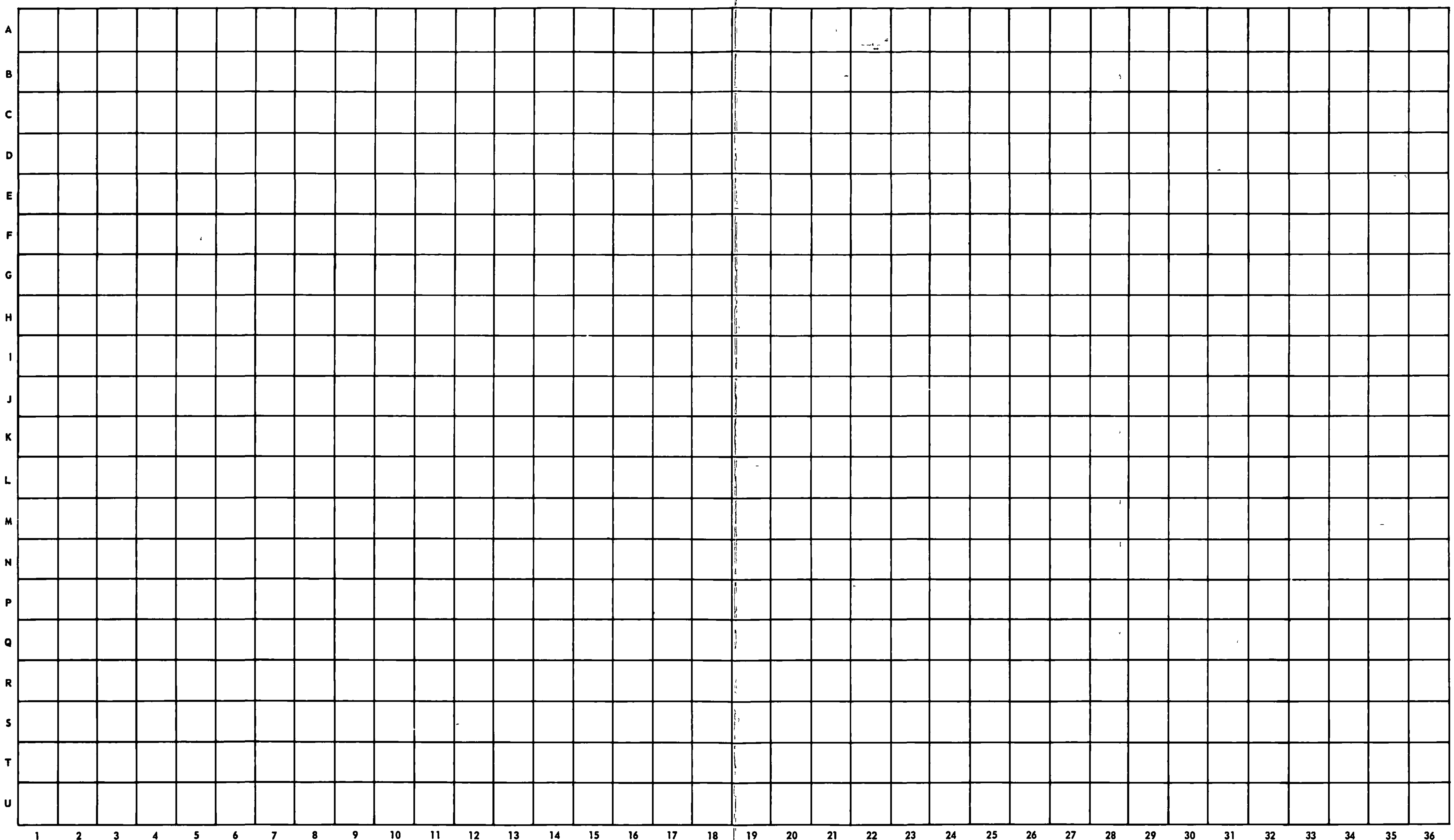
Once the spacecraft components have been plotted for a given target point, the α, β grids are summated using the orthogonal grid overlay shown in Figure B-3. This overlay divides the α, β grid into 720 equal areas, or solid angles, with respect to the target point. Cumulative shielding is determined for each of the 720 solid angles based on both normal and slant height areal density. Results are plotted as areal density versus percent of solid angle less than this value as shown in Figure 8.8-5.

An additional feature of the item by item approach is that it provides a tool for near optimum location of additional shielding should the spacecraft inherent shielding be inadequate.

Radiation "windows" become obvious if cummulative areal density for each solid angle is plotted on Figure B-3. The weight required for a unit increase in areal density for a given solid angle is a function of the intercept area of the solid angle with the spacecraft surface where the additional shielding is added. Intercept areas for each of the solid angles represented in Figure B-3 can be determined from a projection of the alpha beta grid overlay onto a true scale layout of the proper spacecraft surfaces.

To determine the additional shielding, material is first added to the thinnest sections. Material is added to a thinner section before being added to a section of greater thickness. This procedure is continued incrementally until

ALPHA BETA GRID OVERLAY FOR 720 EQUAL SOLID ANGLES



the desired amount of additional dose attenuation is achieved.

Weight of additional shielding is determined from the projected areas and the required pound per square foot increments. The physical location of the additional shielding may be determined either analytically or graphically from the true scale layout.

In this study, approximately 150 items of hardware and equipment constituting 60 percent of the spacecraft weight were evaluated. The items evaluated were those which contribute significant shielding and/or total weight; thus it is believed that the remaining weight, although definitely providing additional shielding, will not greatly change the results obtained in Section 8.8.

Some of the results obtained in Section 8.8 are based on the theoretical addition of shielding at the spacecraft outer moldline. Further optimizations to include selection of the best surfaces for addition of shielding and consideration of multiple target points and the relative effects of projected area and inherent shielding level in choosing which sections to increase are properly the subject of future investigations.

The techinques discussed are readily extendable to include the effect of installation order and variation in the radiation attenuation characteristics of various materials. In addition, the majority of the operations involved are adaptable to mechanization through digital computer programing.

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